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PRELIMINARY AIRWORTHINESS EVALUATION OF A NATIONAL AERONAUTICS & SPACE ADMINISTRATION AUTOMATED STALL WARNING SYSTEM FOR AN OV-1 AIRCRAFT

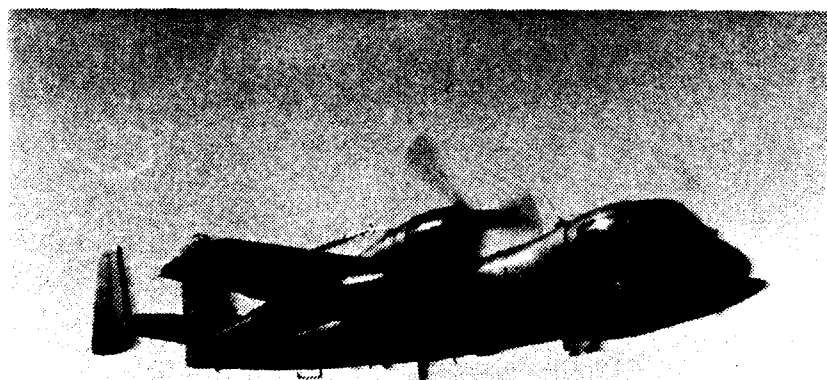USAAEFA

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JULY 1984

FINAL REPORT

UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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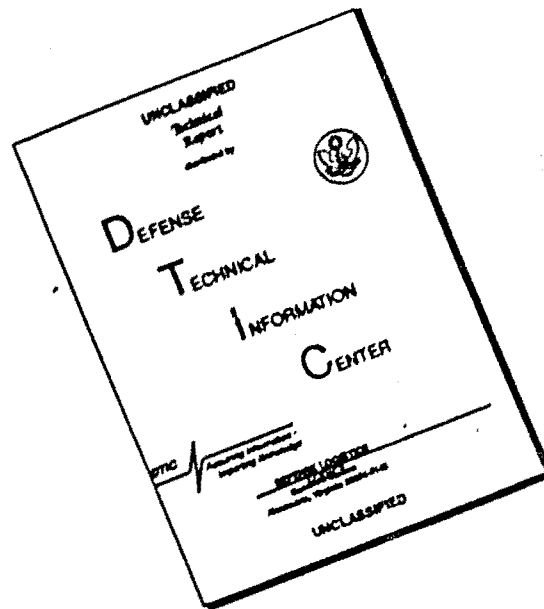
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REPORT DOCUMENTATION PAGE		READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER USAAEFA PROJECT NO. 81-06-1	**2. GOVT ACCESSION NO.** AD-A152 010	**3. RECIPIENT'S CATALOG NUMBER**
4. TITLE (and Subtitle) PRELIMINARY AIRWORTHINESS EVALUATION OF A NATIONAL AERONAUTICS & SPACE ADMINISTRATION AUTOMATED STALL WARNING SYSTEM FOR AN OV-1 AIRCRAFT		**5. TYPE OF REPORT & PERIOD COVERED** FINAL REPORT 5 Dec 83 - 1 May 84
		6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s) DONALD L. UNDERWOOD RICHARD S. ADLER		**8. CONTRACT OR GRANT NUMBER(s)**
9. PERFORMING ORGANIZATION NAME AND ADDRESS US ARMY AVN ENGINEERING FLIGHT ACTIVITY EDWARDS AIR FORCE BASE, CA 93523-5000		**10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS** EN1Y0197
11. CONTROLLING OFFICE NAME AND ADDRESS US ARMY AVIATION SYSTEMS COMMAND 4300 GOODFELLOW BOULEVARD ST. LOUIS, MO 63120-1798		**12. REPORT DATE** July 1984
		13. NUMBER OF PAGES 66
14. MONITORING AGENCY NAME & ADDRESS*(If different from Controlling Office)*		**15. SECURITY CLASS. (of this report)** UNCLASSIFIED
		15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT *(of this Report)*

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT *(of the abstract entered in Block 20, if different from Report)*

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OV-1
Automated Stall Warning System
Preliminary Airworthiness Evaluation

20. ABSTRACT *(Continue on reverse side if necessary and identify by block number)*

The US Army Aviation Engineering Flight Activity conducted a Preliminary Airworthiness Evaluation of a National Aeronautics and Space Administration, Automated Stall Warning System (ASWS) for an OV-1 aircraft. An experimental system independent of conventional angle-of-attack sensors, the ASWS incorporates an onboard micro-computer which displays stall warning margin information as a function of computed calibrated airspeed and stall airspeed in real time. The objective of the test was to evaluate the feasibility of an

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SECURITY CLASSIFICATION OF THIS PAGE *(When Data Entered)*

ASWS as installed in the OV-1 and determine compliance of the system with the requirements of MIL-F-8785C. A limited evaluation of the ASWS was conducted at Edwards Air Force Base, California from 5 December 1983 to 1 May 1984. During the test program 33 flights were conducted, for a total of 37.0 hours, of which 29.0 were productive. In all conditions and configurations tested, the system provided greatly improved stall warning margins when compared to the aerodynamic stall warnings of the OV-1 series aircraft. Adequate stall warnings were provided in most flight configurations tested and the requirements of MIL-F-8785C were met in 156 of the 164 stalls evaluated. System hysteresis and the effects of sideslip on the stall warning capability of the system were minimal and acceptable. One deficiency and one shortcoming were noted during the conduct of this test. The failure of the ASWS to accurately predict single engine stall airspeeds was identified as a deficiency. Additionally, the attenuation of the aural warning cue during radio and interphone communications resulted in decreased crew awareness of impending stall and was a shortcoming. The deficiency must be corrected if development continues. The shortcoming should be corrected prior to production.

REPLY TO
ATTENTION OF

AMSAV-E

SUBJECT: Directorate for Engineering Position on the Final Report of USAAEFA
Report No. 81-06-1, Preliminary Airworthiness Evaluation of a National
Aeronautics and Space Administration Automated Stall Warning System
for an OV-1 Aircraft

SEE DISTRIBUTION

1. The purpose of this letter is to establish the Directorate for Engineering
position on subject report. The objectives of this Preliminary Airworthiness
Evaluation was to evaluate the feasability of a NASA/AEFA developed automated
stall warning system on the OV-1 and determine system compliance with the
requirements of MIL-F-8785C, Flying Qualities of Piloted Airplanes.

2. This Directorate agrees with the report conclusions and recommendations as
stated. The deficiency discussed in paragraph 13 and the shortcoming discussed
in paragraph 17 will be addressed in the event this system is selected for pro-
duction and inclusion in all OV-1 aircraft.

3. The need for a stall warning system in the OV-1 series aircraft is well
known. AVSCOM is investigating several alternative systems and intends to
install a stall warning system as part of the OV-1 Block Improvement Program.

FOR THE COMMANDER:

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DANIEL M. McENEANY
Acting Director of Engineering

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TABLE OF CONTENTS

INTRODUCTION

BACKGROUND

1. The necessity for a stall warning system on Army OV-1 aircraft
has been identified. Flight tests have determined that the
aircraft does not provide adequate aerodynamic stall warning cues
in most configurations (refs 1 through 4, app A). In July 1981,
the US Army Aviation Engineering Flight Ac ivity (USAAEFA) was
tasked by the US Army Aviation Research and Development Command
since redesignated the US Army Aviation Sys ems Command (AVSCOM)
(ref 5) to participate with the National Aeronautics and Space
Administration (NASA), Dryden Flight Research Facility (DFRF) in
the design, development and evaluation of an automated stall
warning system (ASWS) tailored for operation on the OV-1. A
similar experimental system independent of conventional angle-of-
attack sensors had been developed at the NASA Ames Flight Research
Center and flight tested on a NASA Cessna 402B aircraft.

TEST OBJECTIVE

2. The objective of the test was to evaluate the feasibil ty of
a NASA/USAAEFA developed automated stall warning system as i istal-
led in the OV-1 and determine compliance of the system with the
requirements of MIL-F-8785C (ref 6).

DESCRIPTION

3. The test aircraft was a standard short wing OV-1C airplane, US
Army Serial Number 60-03748, equipped with AVCO Lycoming T53-L-7
turboshaft engines. A detailed description of the OV-1C airplane
and its mission related equipment is contained in the operator's
manual (ref 7). The external and internal configuration of the
test aircraft were modified by the installation of various test
equipment which is discussed in appendix C.

4. The automated stall warning system was designed and developed
by the Measurement Section of NASA/DFRF. The system incorporated
an onboard micro-computer configured from a combination of commer-
cially available and NASA/USAAEFA designed hardware which computed
and displayed calibrated airspeed and stall airspeed in real time.
Inputs to the computer included the basic aerodynamic parameters
(dynamic and static pressure, and horizontal and vertical accel-
erations); elevator, flap and gear position, engine torques, and
fuel flow (fig. 1, app B). Components of the system consist of
an airborne instrumentation computer system (AICS), electroservo
circuitry, cockpit mounted airspeed/stall speed indicator, and a
portable visual display unit (photo 2, app B). The AICS provided:

analog-to-digital signal conditioning; micro-processors; nonvola-
tile and volatile onboard memory; parallel interface to input/
ouput circuitry; and an aural warning circuitry utilizing a
Votrax SC-01 phoneme synthesizer and audio power amplifier. The
electro-servo circuitry drove the cockpit mounted airspeed/stall
speed indicator through two separate digital-tosynchro converters.
The indicator (photo 3, app B) for visually displaying airspeed
and stall speed was a modified, dual-needle radio magnetic indi-
cator (RMI) with a standard OV-1 aircraft airspeed indicator
dial face installed. Along with the conventional airspeed pointer
there was mounted another pointer with a red triangle to indicate
the stall airspeed. A detailed description of the ASWS and its
associated hardware is contained in reference 8, appendix A, and
in appendix B.

TEST SCOPE

5. The evaluation of the ASWS was conducted in a three phase
program at Edwards Air Force Base, California from 5 December 1983
to 1 May 1984. Phase I of the program consisted of initial soft-
ware design of the stall airspeed algorithm (equation 1, app D)
and determination of the associated coefficients (equation 10,
app D). These tests were conducted by the Measurement Section
of NASA/DFRF, with USAAEFA assistance, utilizing Grumman Aerospace
Corporation OV-1 wind tunnel and handbook data (ref 7, app A)
through analog/ digital computer simulation. Phase II entailed
quantitative inflight development and evaluation of the ASWS,
followed by qualitative flight testing from a human factor engi-
neering viewpoint in Phase III by six other pilots. During the
test program 33 flights were conducted for a total of 37.0 hours,
of which 29.0 were productive. All tests were performed within
the limitations of the operator's manual (ref 7, app A) as
modified by the airworthiness release (ref 9). Table 1 depicts
the specific aircraft flight configurations tested at the condi-
tions shown in table 2. Sergeant Fletcher 150-gallon external
drop tanks were mounted on wing stations 3 and 4 (photo 2, app B)
at takeoff gross weights greater than approximately 13,300 pounds.
Stall performance characteristics of the ASWS were compared with
MIL-F-8785C (ref 6, app A) criteria to determine compliance.

2

Table 1. Configuration Definition

Configuration	Symbol	Landing Gear Position	Flap Position (deg)
Takeoff	TO	down	15
Cruise	CR	up	0
Slow Cruise	SC	down	0
Slow Flight	SF	up	15
Landing	L	down	45

TEST METHODOLOGY

6. Established flight test techniques and data reduction procedures were used during this test program (ref 10). The test methods are described briefly in the Results and Discussion Section of this report. Flight test data during the approach to the stall, stall, and post-stall reaction were recorded as a time-tagged event in nonvolatile memory in the AICS in a digital format, and hand recorded in the cockpit. A list of test instrumentation is contained in appendix C. Test techniques (other than the standard techniques described in the reference), weight and balance, and data reduction techniques are contained in appendix D. Deficiencies and shortcomings are in accordance with the definitions presented in appendix D.

Table 2. Test Conditions

Type of Test	Gross Weight (lb)	Longitudinal[2] Center of Gravity (FS)	Density Altitude (ft)	Engine Torque Pressure (lb/in²)	CG Normal[3] Acceleration (G)	Remarks
Airspeed Calibration	11,820-13,420	159.4-160.1	9200-10,040	22.0-76.0	1.0	Configuration: CR,SF,L
Stall Performance 0 Degrees Flaps	12,380-16,110	156.9-166.2	10,060-14,120	4.0-76.0	1.0-2.7	Configuration: CR,SF Sideslip Effects Single engine stalls (left engine feathered) Configuration: CR
Stall Performance 15 Degrees Flaps	13,060-13,571	159.9-160.1	11,140-11,470	3.0-64.0	1.0-1.5	Configurations: TO,SF
Stall Performance 45 Degrees Flaps	12,520-16,130	156.9-166.2	10,470-13,980	5.0-69.0	1.0-1.5	Configuration: L Sideslip Effects
Qualitative Evaluation	12,380-13,420	158.9-159.2	10,060-10,470		1.0-2.7	Configurations: TO, CR, SF, SC, L Mission Manuevers: Normal takeoffs and landings, power approach, precision landings, nap-of-the-earth operations, aileron rolls, loops, and recovery from near vertical flight.

NOTES:

1Maximum gross weight: 16,600 lb
2Longitudinal center of gravity limits: FS 156.36 to 167.14
3Accelerated stalls conducted using windup turn method.

4

RESULTS AND DISCUSSION

GENERAL

7. A limited preliminary airworthiness evaluation of the NASA/USAAEFA Automated Stall Warning System was conducted to determine the concept feasibility of the system as installed in an OV-1C aircraft. Dual and single-engine stall performance as well as the effects of system hysteresis and sideslip on the operating characteristics of the system were evaluated at the configurations and conditions listed in tables 1 and 2. The system provided greatly improved stall warning margins when compared to the aerodynamic stall warnings of the OV-1 series aircraft. Adequate stall warnings were provided in most flight configurations tested and the requirements of MIL-F-8785C were met in 156 of the 164 stalls evaluated. System hysteresis and the effects of sideslip on the stall warning capability of the system were minimal and acceptable. One deficiency and one shortcoming were noted during the conduct of this test. The failure of the ASWS to accurately predict single engine stall airspeeds was identified as a deficiency. Additionally, it was qualitatively determined that the attenuation of the aural warning cue during radio and interphone communications resulted in decreased crew awareness of impending stall and was identified as a shortcoming. The deficiency must be corrected if development continues. The shortcoming should be corrected prior to production.

STALL WARNING SYSTEM DEVELOPMENT

8. Configuration of the ASWS with the initial software design of the stall airspeed algorithm (equation 1, app D) and determination of the associated coefficients (equation 10, app D) were conducted utilizing Grumman Aerospace Corporation OV-1 wind tunnel and handbook data (ref 7, app A) through ground based analog/digital computer simulation. The predetermined aural stall warning margin airspeed was set as the greater of 107.5 percent of the calculated stall airspeed (V_S) or 7.5 knots (kt) plus the calculated stall airspeed. This was established as an average of the specification stall warning margin for the power approach (most critical) configuration as specified in paragraph 3.4.2.1.1.1 of MIL-F-8785C. Additionally, the ASWS indicator instrument error and aircraft ship's airspeed system position error were input to the AICS with the resultant ASWS airspeed/stall speed indications being calibrated airspeeds. Therefore, airspeeds presented in this discussion and data presented in appendix E are calibrated airspeeds.
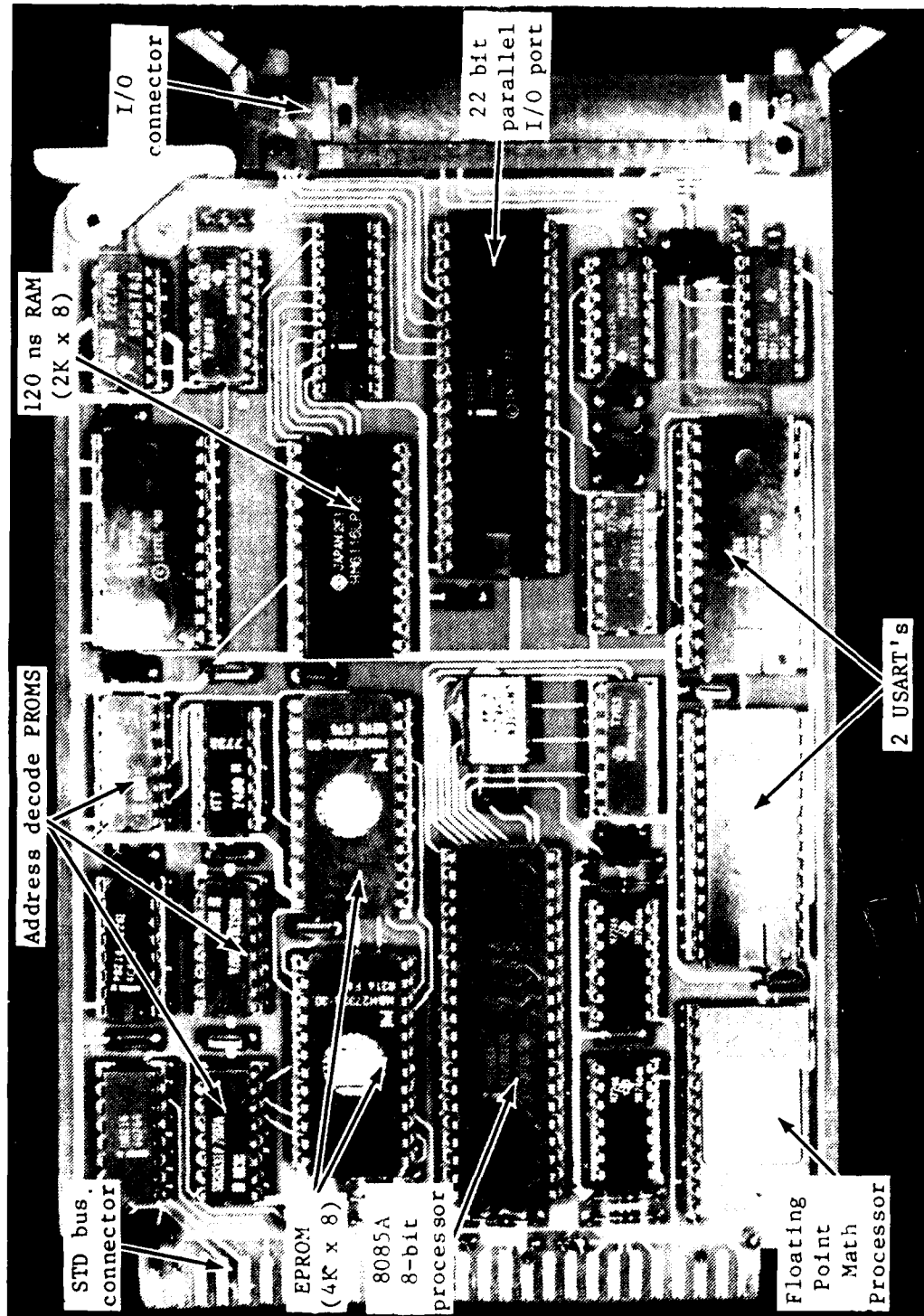
Photo 6. Single-Board Microcomputer

I/O connector

22 bit parallel I/O port

120 ns RAM (2K x 8)

2 USART's

Address decode PROMS

STD bus connector

EPROM (4K x 8)

8085A 8-bit processor

Floating Point Math Processor

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a.   Single-board processor

b.   Nonvolatile electrically erasable programmable read only memory (EEPROM) board

c.   Parallel interface board

d.   Analog-to-digital converter board

e.   Voice synthesizer and audio power amplifier board

Single-Board Processor:

4.   The single-board microcomputer (photo 6) is a fully flight-qualified board designed and built by DFRF on a six-layer printed circuit board.   The printed circuit board was designed so that the STD bus signals are presented at one edge of the board while the special purpose input/output (I/O) signals are presented on the opposite edge. This is a standard approach allowing a more optimum segregation of the two types of signals.   The board contains 24 integrated circuits, as well as an assortment of miscellaneous support circuitry.

5.   From the block diagram of the single-board microcomputer (fig. 2), the following points can be ascertained.   The 8085A microprocessor has its lower 8 bits of address/data lines (AD0-AD7), as well as its control lines (Read, Write, clock, IO/M, Address Latch Enable, and Reset) inputted to an 8155 integrated circuit.   The 8155 provides two 8-bit and one 6-bit parallel input/output ports (used to interface the 8085A to the digital to synchro converters), 256 bytes of static radom access memory (RAM), and a 14-bit programmable timer for the board. Two 2732 erasable programmable read only memory (EPROM) integrated circuits provide the board with 8 kbytes of program memory, with the remaining 2 kbytes of RAM memory being provided by an HM6116LP-2 chip.

6.   Two Universal Synchronous/Asynchronous Receiver Transmitters are provided by two 8251 chips with their programmable baud-rate generators being supplied by two of the three 16-bit timers of an 8253 circuit.   The combination of the 8251s and the 8253 circuits gives two synchronous or asynchronous ports with adjustable baud-rates from 0-19,200 baud for asynchronous modes, and from 0-64,000 baud for synchronous modes.   A 75189 integrated circuit provides the input interface to the 8251s from the RS-232C terminal devices, and a 75188 integrated circuit (IC) handles the output interface from the 8251s to the RS-232C terminal devices.
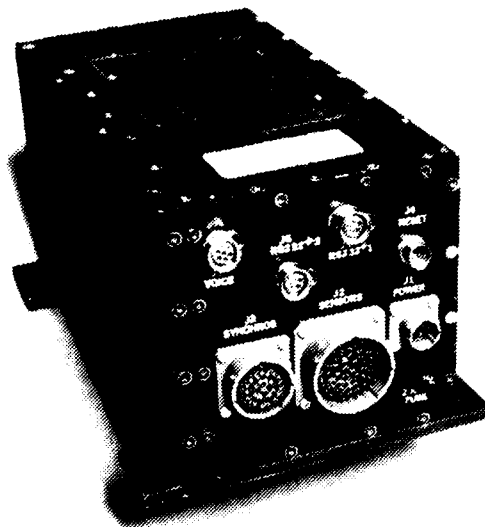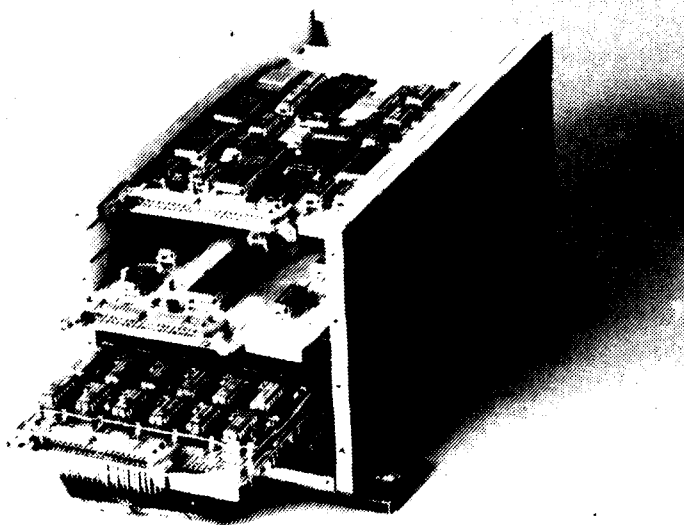
Photo 4.   AICS Box (Front View)
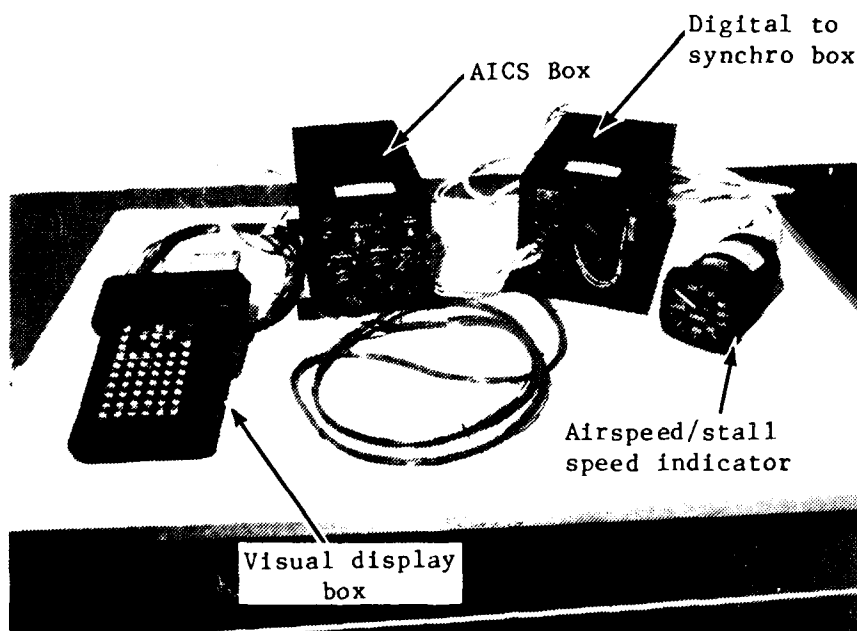

Photo 5.   AICS Box (Internal View)

Photo 2.   Stall-Speed Warning System
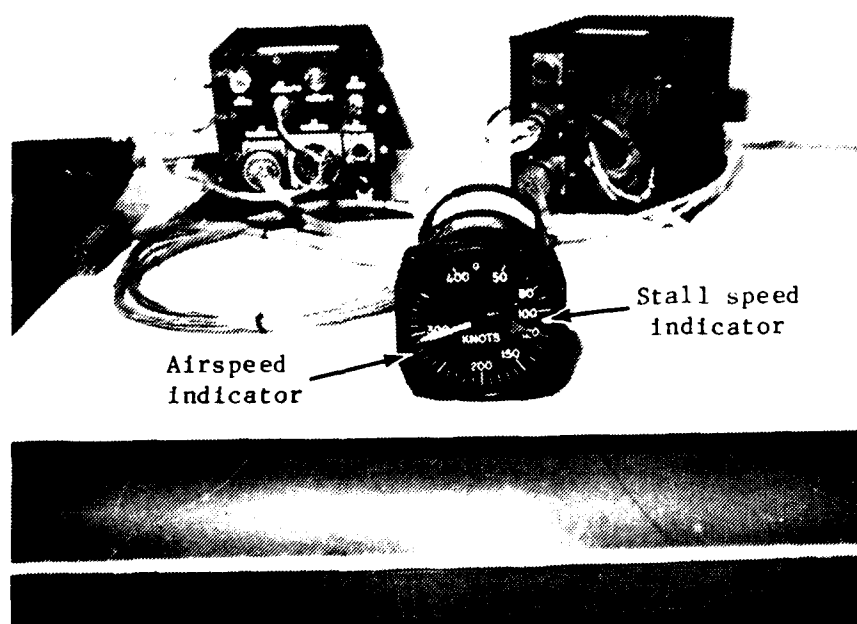

Photo 3.   Cockpit-Mounted Indicator

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Cockpit
indicator

High power synchro
driver box

12 bit digital/synchro
converter/amplifier

12 bit digital/synchro
converter/amplifier

Airspeed
Synchro

Stall
speed
Synchro

14 lines

Input/output
terminal

Portable Ascii
display unit

Aircraft Parameters

Engine torques
Flap position
Elevator position
Airspeed ΔP
Pressure altitude
Longitudinal acceleration
Vertical acceleration
Fuel weight
Landing gear

Audio signals to
aircraft's intercom system

RS-232

Airborne Microcomputer

Processor Board
• 8085A 8 bit μ-processor
• 8231A float point processor
• 22 bits parallel I/O
• 2 USARTS
• RAM, PROM
• Program timers

EE Ram Board
• 4K bytes (2816) E²PROM

Analog Board
• 16 channel (8 ch differential)
• Gain: 1→1000 programmable
• 12 bit A/D converter

Parallel I/O Board
• 48 programmable
  input/output lines

Audio Board
• VOTRAX voice synthesizer
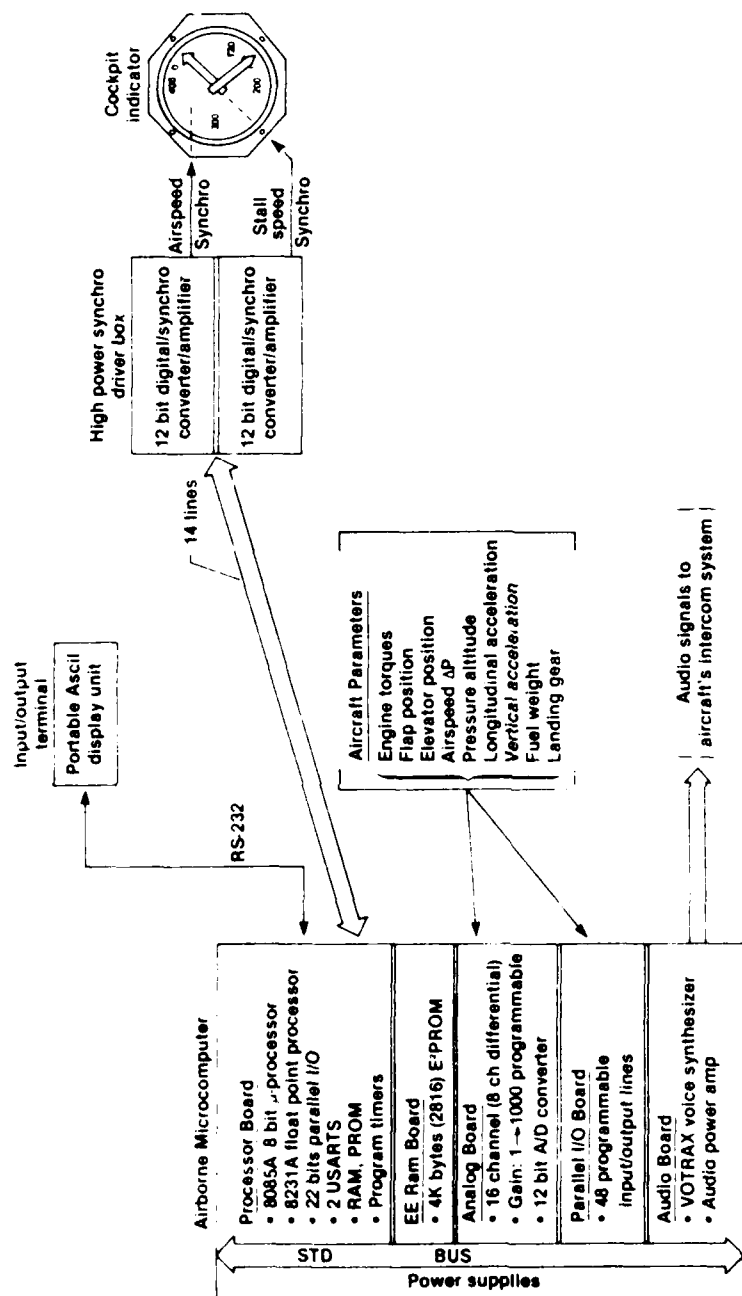• Audio power amp

STD

BUS

Power supplies

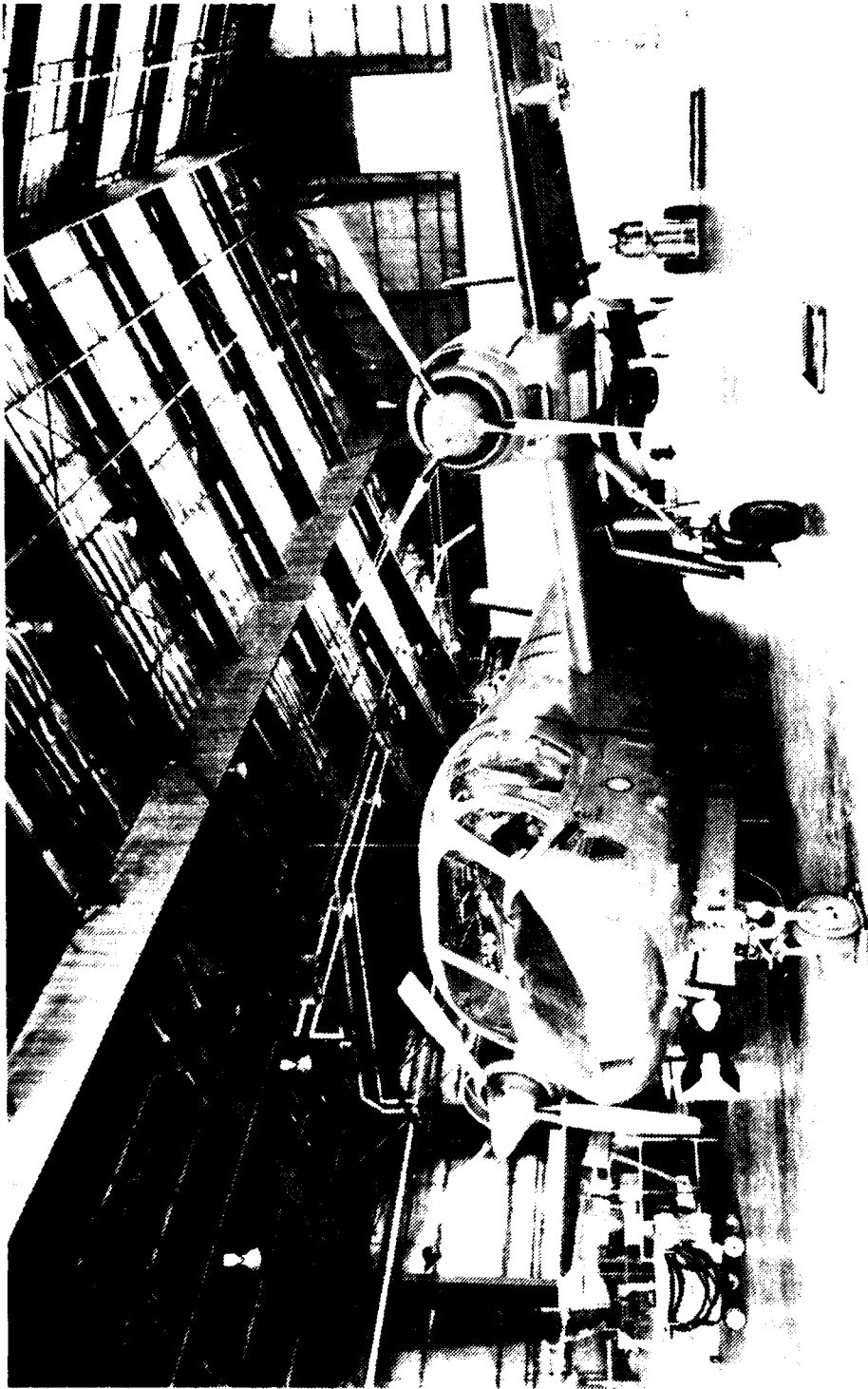Figure 1. Overall Block Diagram of Stall-Warning System

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Photo 1. OV-1C Aircraft

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APPENDIX B. DESCRIPTION

AUTOMATED STALL WARNING SYSTEM

General

1. The automated stall warning system configured for an OV-1
series aircraft (photo 1) was designed and developed by the
Measurement Section of the National Aeronautics and Space
Administration, Dryden Flight Research Facility (NASA/DFRF) with
assistance provided by the US Army Aviation Engineering Flight
Activity (USAAEFA). The system incorporates an onboard micro-
computer configured from a combination of commercially available
and NASA/USAAEFA designed hardware which computes and displays
calibrated airspeed and stall speed in real time. Inputs to the
computer include the basic aerodynamic parameters (dynamic and
static pressure, and horizontal and vertical accelerations) and
other parameters (elevator, flap and gear position, engine
torques, and fuel flow (fig. 1). Components of the system consist
of an airborne instrumentation computer system (AICS), electro-
servo circuitry, cockpit mounted airspeed/stall speed indicator,
and a portable visual display unit (photo 2). The AICS provides:
analog-to-digital signal conditioning; microprocessors; nonvola-
tile and volatile onboard memory; parallel interface to input/
output circuitry; and an aural warning circuitry utilizing a
Votrax SC-01 phoneme synthesizer and audio power amplifier. The
electro-servo circuitry drives the cockpit mounted airspeed/stall
speed indicator through two separate digital-to-synchro con-
verters. The indicator (photo 3) for visually displaying airspeed
and stall speed is a modified, dual-needle radio magnetic indi-
cator (RMI) with a standard OV-1 aircraft airspeed indicator
dial face installed. Along with the conventional airspeed pointer
there is mounted another pointer with a red triangle to indicate
the stall speed.

Airborne Instrumentation Computer System

2. The AICS (photos 4 and 5) is a culmination of NASA/DFRF
effort to standardize much of the system's design for ongoing
and future flight test requirements. The AICS provides a standard-
size box with a single-board microcomputer and a standard back-
plane that will accommodate commercially available or special-
purpose, in-house designed custom boards. The bus used in the
AICS bus is the widely used STD bus developed by Pro-Log
Corporation of Monterey, California.

3. The AICS box contains six slots. As developed and configured
for this project, five slots contain a mixture of commercially
available and in-house designed boards. The complement of boards
used in the AICS box is:

APPENDIX A. REFERENCES

1. Final Report, US Army Aviation Systems Test Activity, Project No. 68-43, *Army Preliminary Evaluation, Preproduction OV-1D (Mohawk)*, March 1970.

2. Final Report, US Army Aviation Systems Test Activity, Project No. 70-03, *Army Preliminary Evaluation II, Production OV-1D (Mohawk), Performance Addendum,* April 1972.

3. Final Report, USAAEFA Project No. 74-20, *HOT BRICK III Evaluation, OV-1D Airplane,* November 1974.

4. Final Report, USAAEFA Project No. 77-22, *OV-1D Airplane with AN/ALQ-147 (XE-1) and (XE-2) Systems Installed,* February 1978.

5. Letter, AVRADCOM, DRDAV-DI, 2 July 1981, amended 30 August 1982, subject: Evaluation of a National Aeronautics and Space Administration Stall Speed Indicator for an OV-1.

6. Military Specification, MIL-F-8785C, *Flying Qualities of Piloted Airplanes,* 5 November 1980.

7. Technical Manual, TM 55-1510-204-10, *Operator's Manual, Army Model OV-1C Aircraft,* 10 April 1979, with changes 1 through 5.

8. Technical Memorandum, NASA TM-84917, *An Automated Stall-Speed Warning System,* February 1984.

9. Letter, AVRADCOM, DRDAV-D, 26 April 1982, Revised 17 May 1982 and 6 October 1983, subject: Airworthiness Release, OV-1C S/N 60-3748, NASA Stall Speed Indicator Evaluation Tests.

10. Flight Test Manual, Naval Air Test Center, FTM No. 104, *Fixed Wing Performance,* 28 July 1972.

RECOMMENDATIONS

23. The deficiency must be corrected if development continues (para 7).

24. The shortcoming should be corrected prior to production (para 7).

CONCLUSIONS

GENERAL

19. The following conclusions were reached upon completion of the Preliminary Airworthiness Evaluation of the ASWS:

    a. In all conditions and configurations tested, the ASWS provided greatly improved stall warning margins when compared to the aerodynamic stall warnings of the OV-1 series aircraft tested (para 17).

    b. System hysteresis and the effects of sideslip on the stall warning capability of the system were minimal and acceptable (para 16).

    c. One deficiency and one shortcoming were identified during the conduct of these tests (para 7).

DEFICIENCY

20. The failure of the ASWS to accurately predict single engine stall airspeeds (para 13).

SHORTCOMING

21. Attenuation of the aural warning cue during radio and interphone communications resulted in decreased crew awareness of impending stall (para 17).

SPECIFICATION COMPLIANCE

22. The ASWS met all the stall warning requirements of the specification, MIL-F-8785C, in 156 of the 164 stalls evaluated.

variations in indicated angle-of-attack of the wing with sideslip and, therefore, are sensitive only to pitot-static position errors caused by air flow interference from the nose of the aircraft which were not fully evaluated during this test. System hysteresis and the effects of sideslip on the stall warning capability of the system were minimal and acceptable.

HUMAN FACTORS

17. Qualitative assessments of the ASWS design, functional use, and stall warning presentations were conducted concurrently with all other developmental tests specified in table 2. Additionally, six other pilots (three of which were not OV-1 qualified), two flight test engineers, and a flight surgeon qualitatively evaluated the ASWS capabilities while performing a total of 108 stalls as well as normal mission maneuvers and aerobatic flight at the various conditions specified in table 2. In all conditions and configurations tested, it was determined the ASWS provided greatly improved stall warning margins when compared to the aerodynamic stall warnings of the OV-1 series aircraft tested. By providing a useful and consistently reliable stall warning indications, the ASWS enabled the pilot to obtain maximum aircraft performance while reducing normally high workloads during takeoffs, approaches, landings, and high-g maneuvering flight. The aural warning cue supplied through the ASWS further provided a clear and succinct "heads-up" warning of impending stall with sufficient time to enable the pilot to react prior to reaching the actual stall. During radio and interphone communications, however, the volume of the aural warning cue was significantly attenuated and, therefore, decreased crew awareness of impending stall. The attenuation of the aural warning cue during radio and interphone communications is a shortcoming.

PITOT-STATIC SYSTEM CALIBRATION

18. The pitot-static position error of the ASWS and the standard ship's system was determined at the conditions presented in table 2 using the calibrated pace method. Due to the capabilities of the ASWS, the initial position error data for the ASWS were input to the system in the form of a second-order polynominal, with the resultant ASWS airspeed indications being in knots calibrated airspeed (KCAS). The residual error, as shown in figures 7 through 12, appendix E, is the result of the optimization of the second-order curve fit used in the position error correction. The position error characteristics of the ship's airspeed system are essentially unchanged from those presented in the operators manual.

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15-Degree Flap Position

14. Takeoff and slow flight configuration stalls were conducted at the conditions specified in table 2. The results are presented in figure 5, appendix E. Although there was no clear aerodynamic warning observed during the 12 unaccelerated and six accelerated (1.5 g) stalls evaluated, the ASWS did provide consistent and reliable stall warning indications with sufficient time to enable the pilot to react prior to reaching the actual stall. All but two of the 18 stalls performed met the requirements of MIL-F-8785C. The two accelerated stalls failing to meet the specification requirements had reduced aural stall warning margins of less than 3 kts, but are considered acceptable due to the reliable prediction of the actual stall airspeed. The stall warning provided by the ASWS was satisfactory in the takeoff and slow flight configurations.

45-Degree Flap Position

15. Thirty-five unaccelerated and twenty-three accelerated stalls in the landing configuration were evaluated at the conditions specified in table 2. The results are presented in figure 6, appendix E. In this configuration the aerodynamic prestall warning was virtually nonexistent and the only warning was furnished by the ASWS. For all conditions tested the aural warning system provided a consistent and reliable stall margin with suficient warning to avoid unintentional stall with a maximum error in the predicted stall airspeed of less than 1.5 kts for 56 of the 58 stalls evaluated. The specification requirements of MIL-F-8785C were met in 57 of 58 stalls tested, with the one stall having less than 3.5 kts aural warning margin. The stall warning provided by the ASWS was satisfactory in the landing configuration.

System Hystesis and Sideslip Effects

16. Hysteresis errors in the ASWS were effectively eliminated through the electro-servo circuitry interfacing the digital-to-synchro modules of the airspeed/stall speed indicator to the AICS which updated the ASWS indicator approximately 20 times per second. The effect of sideslip on the stall warning system was evaluated in conjunction with tests conducted in the cruise and landing configurations as specified in table 2. Stalls were performed with sideslip angles of up to 10 degrees right and 4 degrees left (ball-centered). Results of the evaluation indicated no significant variation of the aircraft coefficient of lift at the stall for the various sideslip angles tested. Additionally, the operational characteristics of the ASWS are independent of

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STALL WARNING SYSTEM PERFORMANCE

Zero-Degree Flap Position

12. Dual engine unaccelerated and accelerated stalls and single engine unaccelerated stalls were conducted at the conditions specified in table 2. The results are presented in figure 4, appendix E. Of the 84 dual engine stalls evaluated the aerodynamic prestall buffet occurred from 12 to 14 kts prior to the actual stall airspeed and was virtually indistinguishable from light turbulence and, therefore, did not provide adequate warning. In 67 of these stalls the ASWS provided the pilot reliable stall margin presentation with a maximum error of +1.5 kts, while the remaining 17 stalls performed showed a maximum error of no more than 3.8 kts. For all conditions tested, the predetermined aural stall warning occurred within sufficient time to enable the pilot to react prior to reaching the actual stall and met the requirements of MIL-F-8785C in 80 of the 84 dual engine stalls evaluated. In the cruise and slow cruise configurations, the dual engine unaccelerated and accelerated stall warning capabilities of the ASWS are satisfactory.

13. During single engine unaccelerated stalls in the cruise configuration there was no well defined indication of an aero-dynamic prestall warning. The only warning was furnished by the predicted stall airspeed presentation in the ASWS cockpit indi-cator with the associated predetermined aural warning margin. Although two of the four stalls evaluated met the requirements MIL-F-8785C the single engine stall data did not conform to the general trend of all other data presented in figure 1, appendix E (increasing lift coefficients with increasing power). This was probably due to the engine torque averaging method (equation 6, app D) used in the determination of the aircraft predicted stall airspeed. For the tests conducted at power settings in excess of 40 psi of torque the warning margin was reduced to less than 2.5 kts with a maximum error in the predicted stall airspeed of approximately 4 kts. Although the ASWS provided sufficient information to prevent inadvertent single engine stall entry at low power settings (less than 40 psi of torque), the inaccurate determination of the predicted stall airspeed and inadequate aural warning was considered unacceptable. Due to the increased possibility of spin entry with large power assymetry at the time of stall the failure of the ASWS to accurately predict single engine stall airspeeds is a deficiency.

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9. Subsequent modification of the coefficients to the stall speed algorithm and evaluation of the stall performance capabilities of the ASWS were performed during 164 dual engine power-on and off unaccelerated (1 g) and accelerated (1.3 to 2.7 g) stalls and single engine power-on unaccelerated stalls at the configurations and conditions listed in tables 1 and 2. Dual engine unaccelerated and accelerated stalls were conducted in coordinated (ball-centered) flight at the required bank angle and power setting from a trimmed airspeed of approximately 20 percent greater than the anticipated stall airspeed specified in the operator's manual (ref 7, app A) for the configuration tested. Single engine stalls were performed in the cruise configuration by stabilizing and trimming the aircraft in ball-centered level flight at the specified best single engine climb airspeed for the cruise configuration (ref 7, app A). A 1 kt/sec or less deceleration for unaccelerated stalls and a 2 kt/sec or less deceleration for accelerated stalls was maintained until the stall occurred.

10. In all configurations tested, aerodynamic warning of impending stall was either virtually nonexistent or occurred too far above the actual stall speed to provide adequate warning. Normal and accelerated dual engine stalls were defined by a center of gravity (cg) normal acceleration decrease. Stalls during single engine operations with the left (critical) engine feathered and the right engine operating from 20 psi torque to military rated power were achieved prior to the minimum control airspeed and were characterized by high pitch and roll rates to the left. The ASWS provided greatly improved stall warning margins and the requirements of MIL-F-8785C were met in 156 of the 164 stalls conducted.

11. Data recorded during the approach to the stall, stall, and, post stall were analyzed to determine the coefficients required to provide a minimum of error in predicted stall airspeed and therefore, provide stall warning margins within the requirements of MIL-F-8785C. Since the values of the coefficients (equation 10, app D) were dependent upon the flap position selected, three sets of these coefficients were required for the three possible flap settings (0, 15, and 45 degrees). Each set was optimized based on various gross weight, cg, and engine thrust conditions for the coefficient of lift at stall using a 2nd-order polynomial regression. Data sets utilizing the above technique and their associated coefficients are shown in figures 1 through 3, appendix E.
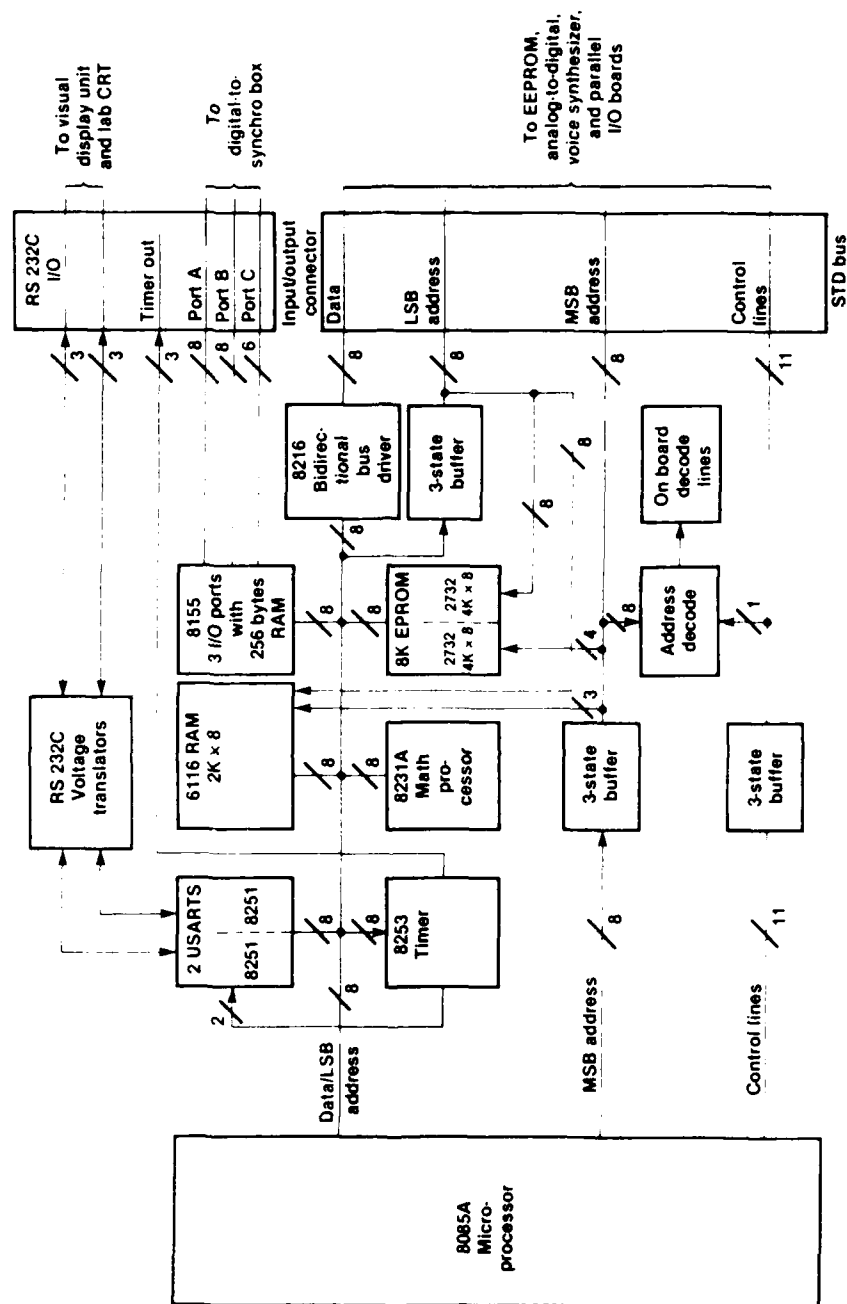
6

Figure 2. Block Diagram of Single-Board Microcomputer

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7. All on-board address decoding is provided by three 82S131 programmable read only memory (PROM) (512 x 4 each). The use of PROM address decoders for all the I/O port address selection results in a significant reduction in chip count and circuit complexity over the more conventional discrete logic with switch-selectable addresses. The penalty, however, is that the address decoding can only be altered by programming a new PROM, and is slower than the discrete logic form.

8. The 8231A floating-point processor is interfaced to the 8085A through the lower 8 bits of the address/data bus (AD7-AD0), as well as the control lines for Read and Write, clock, chip select (from the 82S131 PROMs), and the least significant address line, A0. This address line, in conjunction with Read and Write signals, tells the 8231A what types of operations (e.g., whether to enter data onto its stack, or whether the data bus is to be interpreted as a command from the 8085A, etc.) it will be doing. The 8085A pa es data and commands (what arithmetic operations to do) to the 8231A via the 8-bit data bus, and internally places the data into a stack that is oriented as 8 by 16 bits (or 4 by 32 bits). This stack size allows the 8085A to put eight single-precision, fixed-point operands in it or four double-precision or floating-point operands in the stack. (The 8231A stores a single-precision fixed-point value in a 16-bit word with the most significant bit as the sign bit; a double-precision fixed-point value is stored as a 32-bit word with the MSB being the sign. The floating-point format utilizes a 32-bit word with the least significant 24-bits being the mantissa and the most significant 8 bits representing the exponent. Thus the range of values represented in this way is $\pm(2.7 \times 10^{-10}$ to $9.2 \times 10^{18})$.

Electrically Erasable Programmable Read-Only Memory:

9. The EEPROM board allows storage of coefficients and data from the sensors during the flight-test phase of the project. After the flight, the data can be processed for post flight analysis. The EEPROM board is a commercially available board that uses Intel 2816 nonvolatile EEPROM ICs. The board, when fully populated with eight 2816 chips will allow for 16 kbytes of memory; however, in the stall-speed warning project only 2 kbytes are being used.

10. The EEPROM board is essentially treated by the 8085A microprocessor as an off-board pseudo-RAM memory interfaced through the STD bus. The reason it is viewed as psuedo-RAM is because the Write operation to the board is longer than that for normal RAM; hence, Wait states are required by the 8085A (this requires that the 8085A do nothing during this Wait period). Also, because of the nature of the Write electronics on the EEPROM board, a

Read operation immediately after a Write operation is not advised since the Write electronics take several microseconds to settle.

Parallel Interface Board:

11. The parallel interface board is also a commercially available board. The board selected for this project uses two 8255 ICs to provide 48 programmable I/O lines. The board interfaces to the single-board processor via the STD bus, and interfaces to the external world via a locking edge connnector on the opposite end of the board. This connector interfaces the two engine fuel-used totalizers to the board.

Analog-to-Digital Board:

12. The analog-to-digital board is also a commercially available board (see photo 7). It utilizes a precision instrumentation amplifier giving programmable gains of from 1 to 1,000. The output of the amplifier is then converted to a 12-bit digital word (in offset binary, two's complement, or straight binary format) at a maximum rate of 25,000 channels/sec. The board is capable of accepting either 16 single-ended inputs or 8 differential inputs. With the addition of two more multiplexer ICs, the board capacity can be doubled for both cases. This project required 9 inputs and thus the board was configured with 16 inputs.

13. The board interfaced to the single-board processor via the STD bus, and interfaced to the analog-sensor signals through a locking edge connector on the opposite end of the board. The board is memory-mapped I/O with selectable base-page address on the board itself. The board uses three consecutive bytes in the memory-mapped space as follows: the first 8-bit byte is written into by the 8085A microprocessor specifying the channel number to be read; the next byte is used by the board to store the least significant 8 bits of the 12-bit converted word; the last byte is used for the most significant 4 bits of the converted word, as well as a "Busy" bit to signify to the processor that a conversion is not yet done.

Voice Synthesizer/Audio Amplifier Board:

14. This board is an in-house design interfacing the processor board to a voice synthesizer and audio power amplifier via the STD bus. The board utilizes a Votrax SC-01 phoneme speech synthesizer which has a repertoire of 64 different phonemes (addressable through a 6-bit data word) and 2 bits governing the pitch of the phoneme. Additional pitch control is obtained by

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using a 5497 binary-rate multiplier. An 8755 (2 kbytes of EPROM and 16 input/output lines) is also on this board. The analog synthesized word, airspeed, is then amplified with an adjustable-gain power amplifier capable of driving approximately 0.5 Watts into an 8-ohm speaker, or driving a higher impedance audio inter-com system (the OV-1C intercom system utilizes a Collins C-1611D which appears as an approximately 150-ohm load).

15. When the processor has determined that the airspeed is within a predetermined percentage of the stall speed, the processor outputs an aural warning to the pilot through this board. The predetermined airspeed is set as the greater of 107.5% of the indicated stall speed or 7.5 knots plus the indicated stall speed. These two values can be increased or decreased in real time through the portable visual display unit (para 19) or terminal. Once amplified, this synthesized message is fed into the pilot's headset alerting him to observe his airspeed. Also, an algorithm is used to increase the frequency at which this aural message is sent. This frequency is based on the difference between the airspeed and stall airspeed values, but will not increase to more than 0.9 times a second.

Digital-to-Synchro Modules

16. Two separate calculated values, one for the indicated airspeed and the other the indicated stall speed, are stored as 12-bit digital values. These two values are outputted to the digital-to-synchro box (photo 8) via the parallel port located on the single-board computer. Both values are outputted on the same 12-bit digital bus; however, the separate trigger signals to each separate synchro board loads the proper board with the proper value at the right time. This reduces the number of lines interfacing the two digital-to-synchro boards to the processor board from 27 to 15.

17. The digital-to-synchro converters latch the 12-bit digital values into a register with the advent of the trigger pulse. Next, the converter transforms the 12-bit signal into the sin and cosine signals of the resolver format. The resolver format signals are then converted into the conventional synchro signals S1, S2, and S3 through the use of an electronic Scott-T filter. These synchro signals are then amplified by highpower linear amplifiers to provide the 3-A peak drive capability needed for some torque receivers (fig. 3). The synchro receivers in the airspeed/stall speed indicator do not need all of this drive capability; however, at the time of box design the synchro types had not been determined.
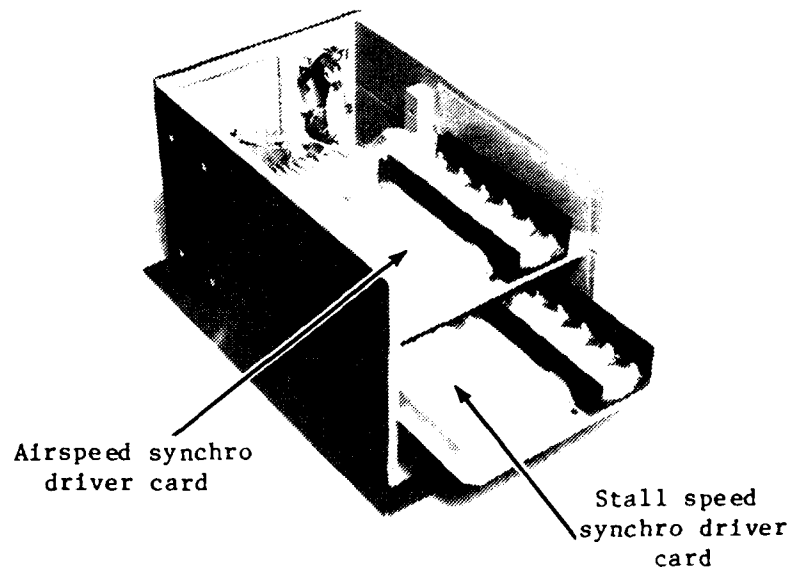
23

Airspeed synchro
driver card

Stall speed
synchro driver
card

Photo 7. Digital-to-Synchro Box

12 Bit A/D
Converter
$(25\ \mu\alpha)$

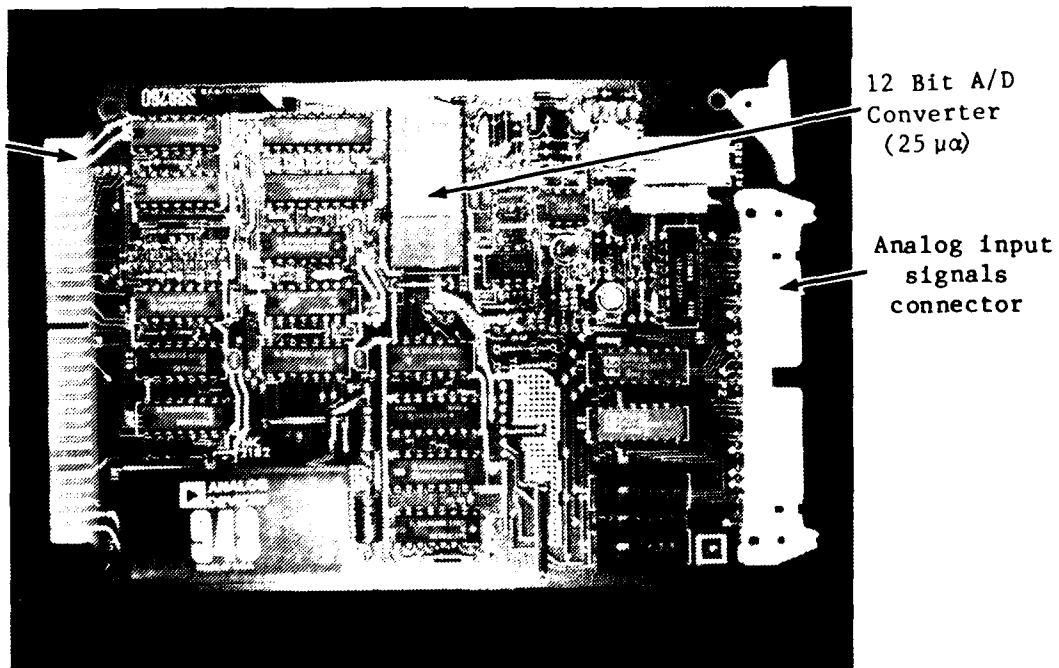Analog input
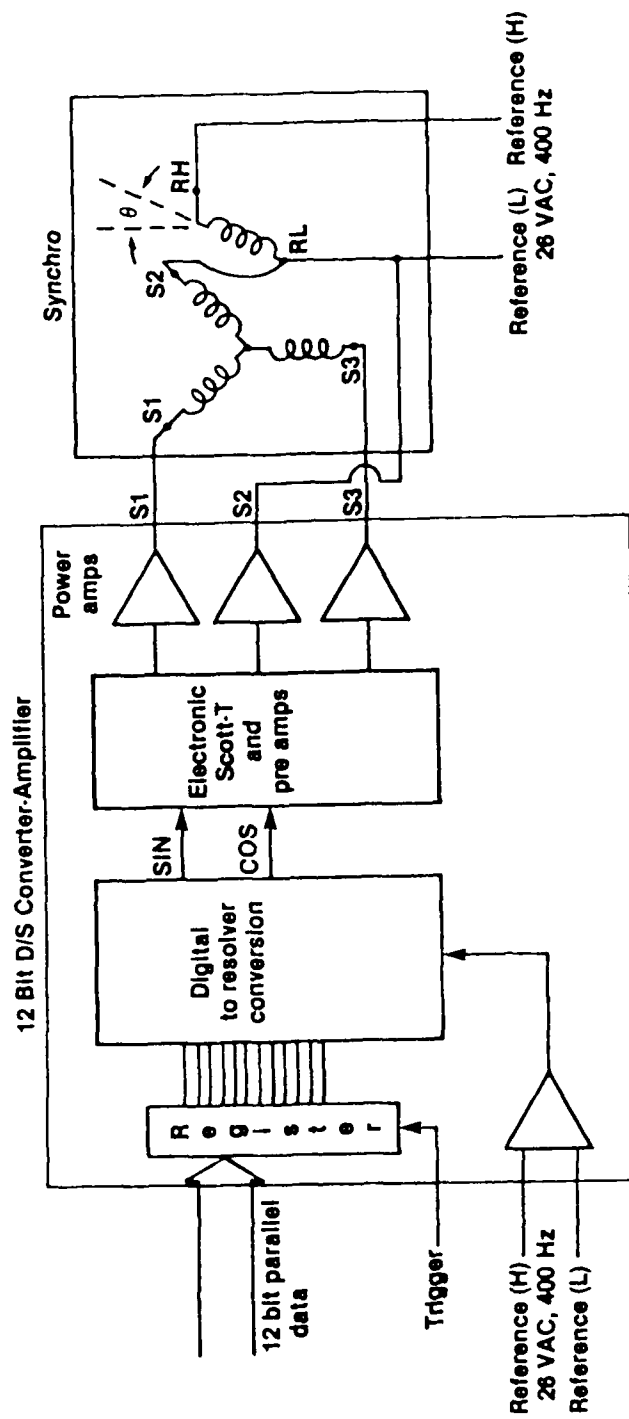signals
connector

Photo 8. Analog-to-Digital Board

Figure 3. Typical Digital-to-Synchro Driver

Portable Visual Display Unit

18. The portable display unit chosen for this project (photos 9 and 10) has the ability to input and output information through a serial RS-232C port located on the single-board processor board. The unit has selectable baud rates to 19,200 baud, a display of 40 characters (set up as two rows of 20 characters per row), and a buffer memory (non-volatile) of 384 lines at 20 characters per line (7.5 kbytes of memory).

19. Software was written for the system to allow the project engineer to load "snapshots" of raw data or processed data into the visual display unit's buffer memory. A sample of the typical display is shown in photos 9 and 10. Although the unit is primarily used as a display unit, it functions as an input device to the processor, thus allowing the operator to change coefficients in the stall-warning software that may be stored in either RAM or EEPROM.

System Software

20. The stall-warning system software is made up of five major sections:

    a.   Initialization routine

    b.   Input data formatting

    c.   Data collection

    d.   Data processing and velocity calculations

    e.   Cockpit indicator drive and aural output

21. The scenario for the first phase of the flight testing is to collect data at predetermined flight conditions. These data would be time-tagged with an event marker that the flight-test engineer would actuate at specific flight conditions. At each time-tagged event, all parameters are loaded into the EEPROM circuits. This data group consists of approximately sixteen 16-bit words. A maximum of 60 separate snapshots would be obtained during the flight, thereby requiring about 2 kbytes of nonvolatile memory storage. Upon completion of the flight, the raw data would be outputted to a printer and analyzed on a larger computing system. The resultant was the determination of the stall speed algorithm coefficients. These coefficients would then be loaded back into the AICS box via the visual display unit and a new flight would be flown to check the validity of the stall-speed calculations.
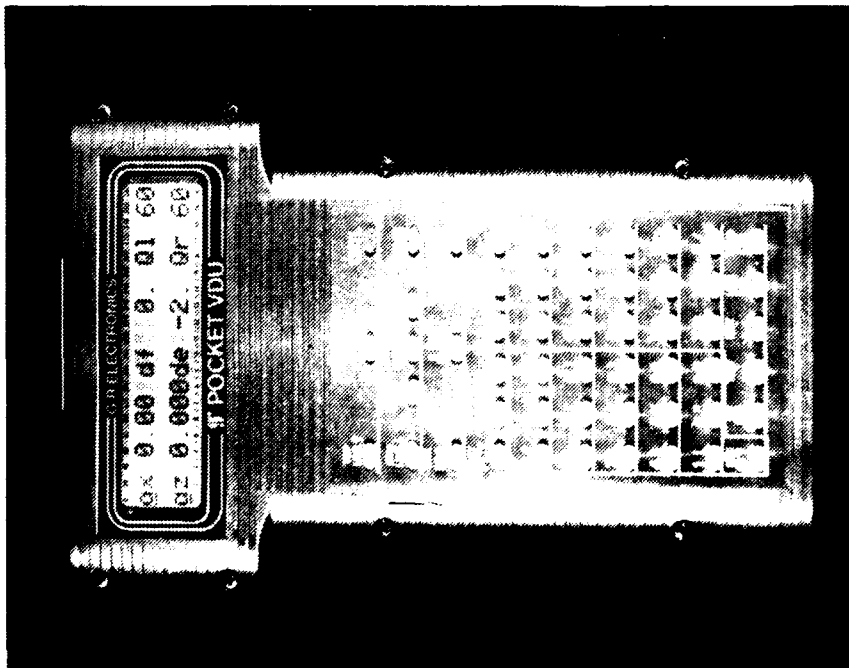
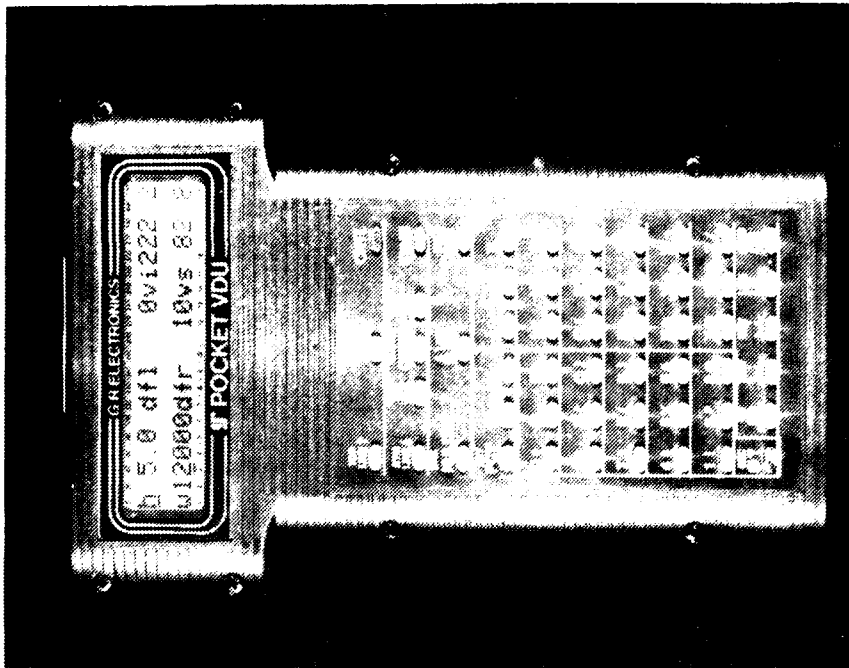Photo 10. Visual Display Unit (Page 2 Format)

Photo 9. Visual Display Unit (Page 1 Format)

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22. During this phase of the flight testing, the visual display unit is also displaying the various parameters (see photos 9 and 10) to the flight-test engineer. Indicated velocity, as well as indicated stall velocity, is shown at this time; however, the validity of the stall speed is only as good as the coefficients used up to that point.

23. Data collection is done in about 10 to 54 msec for the entire set of parameters. This ensures that each parameter is updated at a rate of approximately 20 times per second. Approximately 44 msec of the 54 msec period is necessary for outputting to the display. The visual display is updated once every 0.5 sec, while the cockpit indicator is updated each sample. The routine takes approximately 10 msec to update the sensor calibrations and the synchros. Every 0.5 sec, an additional 44 msec is needed to update the visual display.

24. From the flowchart in figure 4, the portion on the left-hand side is primarily involved with obtaining data, calculating VSIKTS and VIKTS, and displaying them, while the part on the right-hand side is dealing with the entering of the constants and the selecting of options. If no event key is selected, then the system constantly is collecting data at 10-54 msec intervals, calculating the velocities, checking to see if illegal sensor readings or an illegal velocity calculation is attempted (e.g., a negative radical), displaying these values on the indicator, as well as the visual display unit, and outputting the aural warning if necessary. If an event switch is actuated, the values of the parameters are stored in EEPROM, and then the address pointer for the current EEPROM position is updated for the next set of data.

25. If a key-stroke is detected from the visual display unit, the airspeed and stall-speed pointers are driven to 0, and the processor enters a routine to decode the key-stroke entry and perform that function. If an "S" is detected, the processor expects the velocity stall-margin factor to be inputted from the visual display unit. If a "W" is detected, the processor expects the original operating weight (OW) of the aircraft to be entered. A "1" sets the display up for the first page format for displaying to the flight test engineer. A "2" sets up the second page format for display. A "C" followed by two successive "#" entries displays the constants. A "P" takes the contents of the EEPROMs and outputs them to a printer. A "U" configures the display pages for uncalibrated sensor counts display. An "X" configures the display pages for calibrated parameter display. An " ↑" initializes the EEPROM format. An "I" enables the aural warning output while an "L" disables this output. An "*" sets the pilot indicator to the stall margin indicator mode, while a "/" sets the indicator to the normal airspeed, stall speed mode.
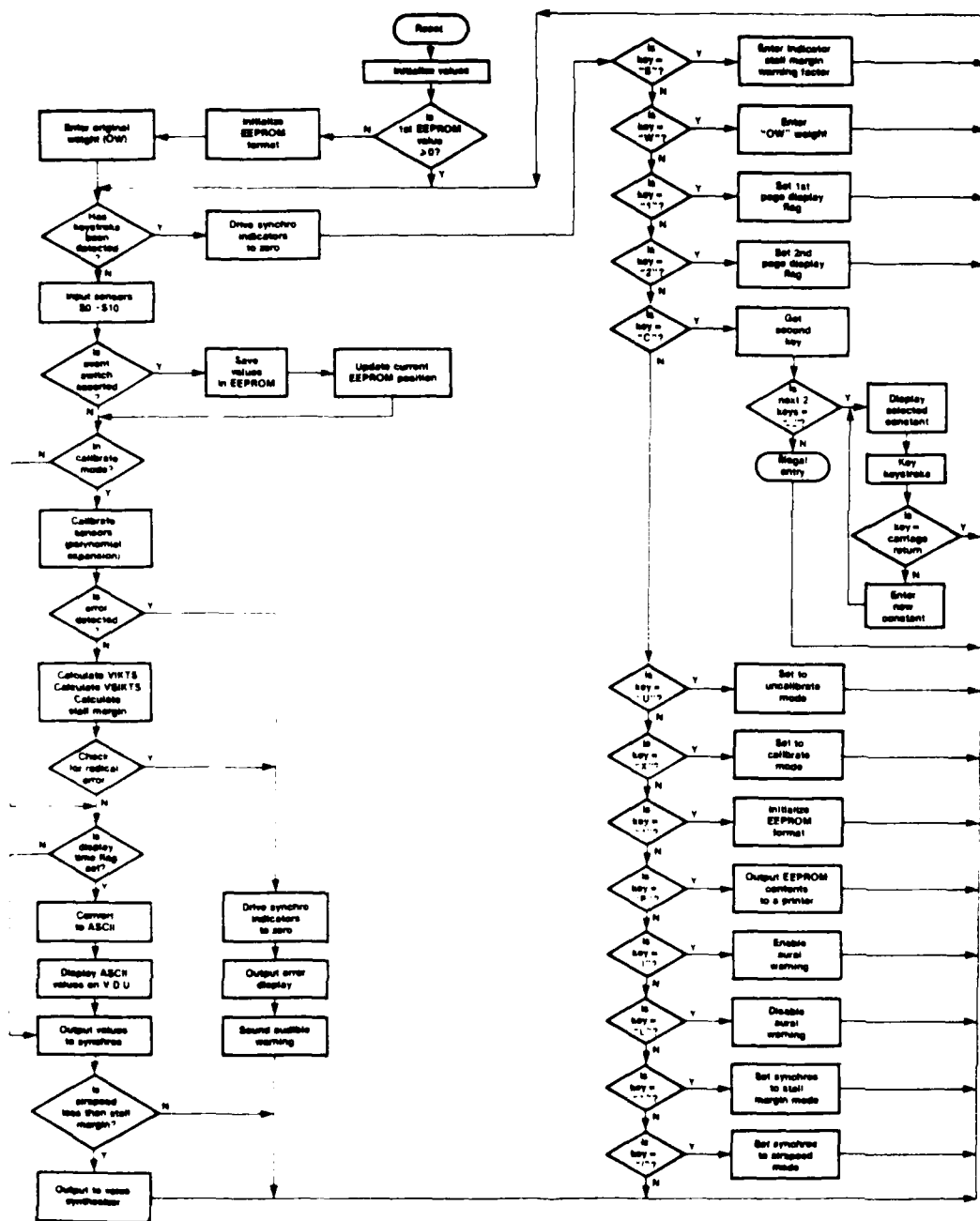
28

Figure 4. Software Operations Flowchart

26. The program will always revert back to the section on taking data and processing it until either a key-stroke from the visual display unit is detected or until an error in calculating is detected or until an error in calculating occurs. These two instances will cause the processor to either decode the key-stroke and perform the task indicated by the key-stroke, or to sound an aural warning and drive the airspeed and stall speed indicator to 0, and display error information on the visual display unit. In either case, the program reverts back to normal data acquisition mode once the keystroke sequence is completed or the error condition is corrected.

APPENDIX C. INSTRUMENTATION

GENERAL

1. Test instrumentation was installed, calibrated and maintained by US Army Aviation Engineering Flight Activity (USAAEFA) personnel. Portions of the test instrumentation served to supply both recorded data for analysis and as input data to the automated stall warning system (ASWS). Since analog-to-digital conversions were performed within the ASWS hardware, data supplied by the test instrumentation (with the exception of fuel used) was in analog form. A specially constructed boom, with a swiveling pitot-static tube and angle-of-attack and sideslip vanes, was installed outboard on the right wing. Figures 1 through 3 present position error corrections for the boom airspeed system in various aircraft and flight configurations. Data were obtained from calibrated instrumentation and displayed or recorded as indicated below.

Pilot Panel

Airspeed (ship's)
Altitude (ship's)
Rate of climb*
Vertical acceleration (sensitive)
Flap position
Engine torque (2)*
Propellor RPM (2)*
Turn and bank*
Event switch

Engineers Panel

Airspeed (boom)
Altitude (boom)
Angle-of-sideslip
Event switch
Angle-of-attack
Calibrated airspeed**
Predicted stall speed**
Outside air temperature**

Visual Display Unit

Predicted stall speed**
Calibrated airspeed**
Altitude
Gross weight**

*Ship's system not calibrated
**Derived parameter

Fuel used (2)
Vertical acceleration (linear)
Longitudinal acceleration (linear)
Flap position
Longitudinal control position
Engine torque (2)

2. Data parameters recorded onboard the aircraft include the following:

Digital Data Parameters

Stall equation coefficients
Event number
Gross weight
Lift coefficient
Predicted stall speed
Calibrated airspeed
Vertical acceleration (linear)
Longitudinal acceleration (linear)
Flap position
Longitudinal control position
Altitude
Fuel used (2)
Engine torque (2)

FIGURE 1
BOOM AIRSPEED CALIBRATION
JOV-1C USA S/N 60-3748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION		AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AIRCRAFT CONFIGURATION
		LONG (FS)	LAT (BL)			
⊡	12910	159.7	0.0	10040	12.0	CRUISE
⊡	13200	158.9	0.0	9420	9.5	CRUISE

NOTE: 1. T-28 PACE METHOD

2. LANDING GEAR RETRACTED, ZERO DEGREES FLAPS

LINE OF ZERO ERROR

NOT FOR HANDBOOK USE

⊙ NO STORES
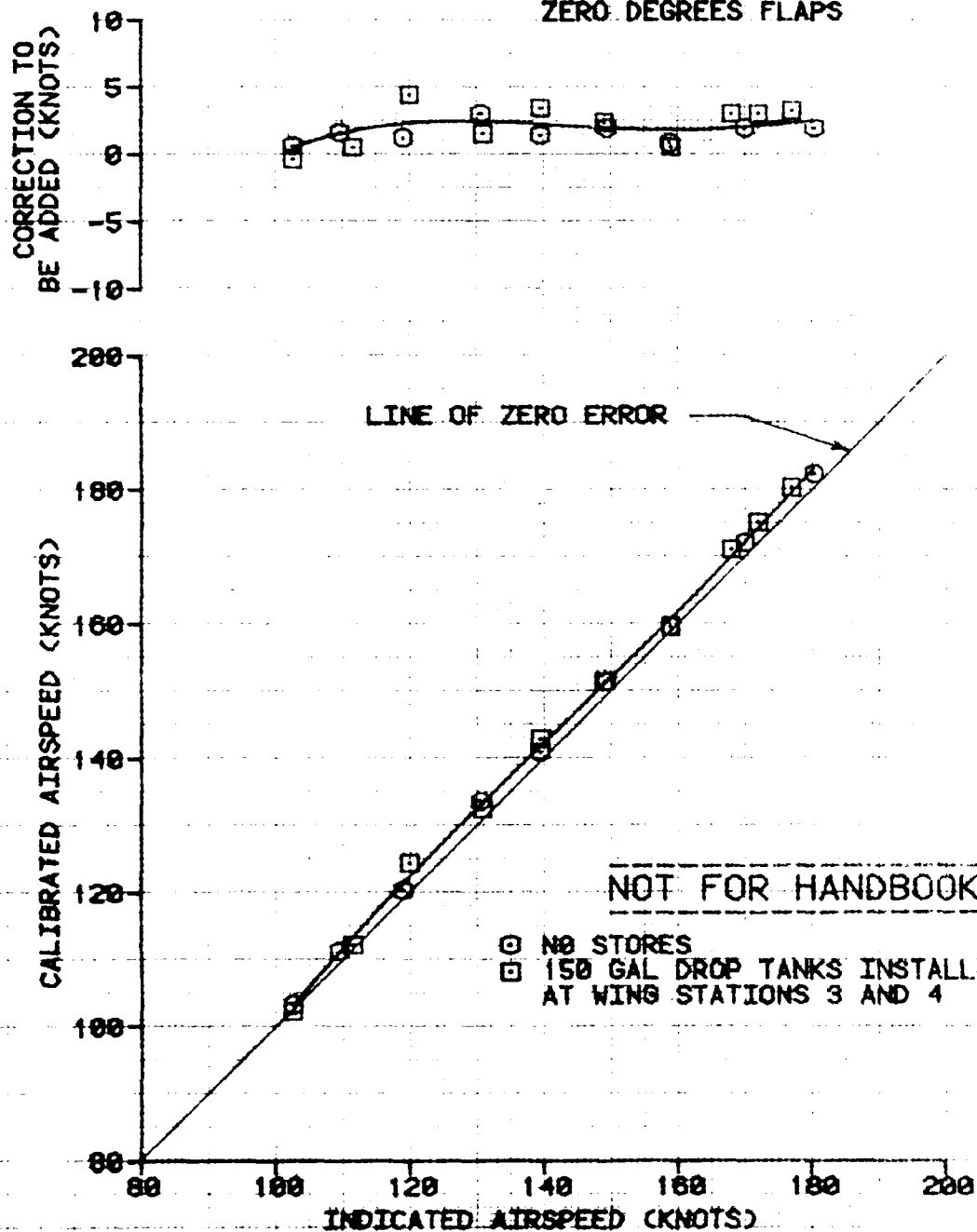⊡ 150 GAL DROP TANKS INSTALLED AT WING STATIONS 3 AND 4

FIGURE 8
AUTOMATED STALL WARNING SYSTEM
STALL AIRSPEED SUMMARY
45 DEGREES FLAPS
JOV-1C USA S/N 60-03748

SYM	AVERAGE GROSS WEIGHT (LB)	AVERAGE LONGITUDINAL CG LOCATION (FS)	AVERAGE DENSITY ALTITUDE (FEET)	AVERAGE VERTICAL ACCELERATION (G)	PROPELLER SPEED (RPM)	LANDING GEAR POSITION
	12980	157.0	11150	1.0	1600	EXTENDED
	12520	156.9	10470	1.5	1600	EXTENDED
	13480	160.0	13980	1.0	1600	EXTENDED
	13440	160.0	13840	1.3	1600	EXTENDED
	15820	160.1	11500	1.0	1600	EXTENDED
	15600	160.0	11600	1.4	1600	EXTENDED
	16130	166.0	11400	1.0	1600	EXTENDED
	15880	166.2	11840	1.3	1600	EXTENDED

NOTE: STALL MARGIN IS DEFINED AS THE DIFFERENCE
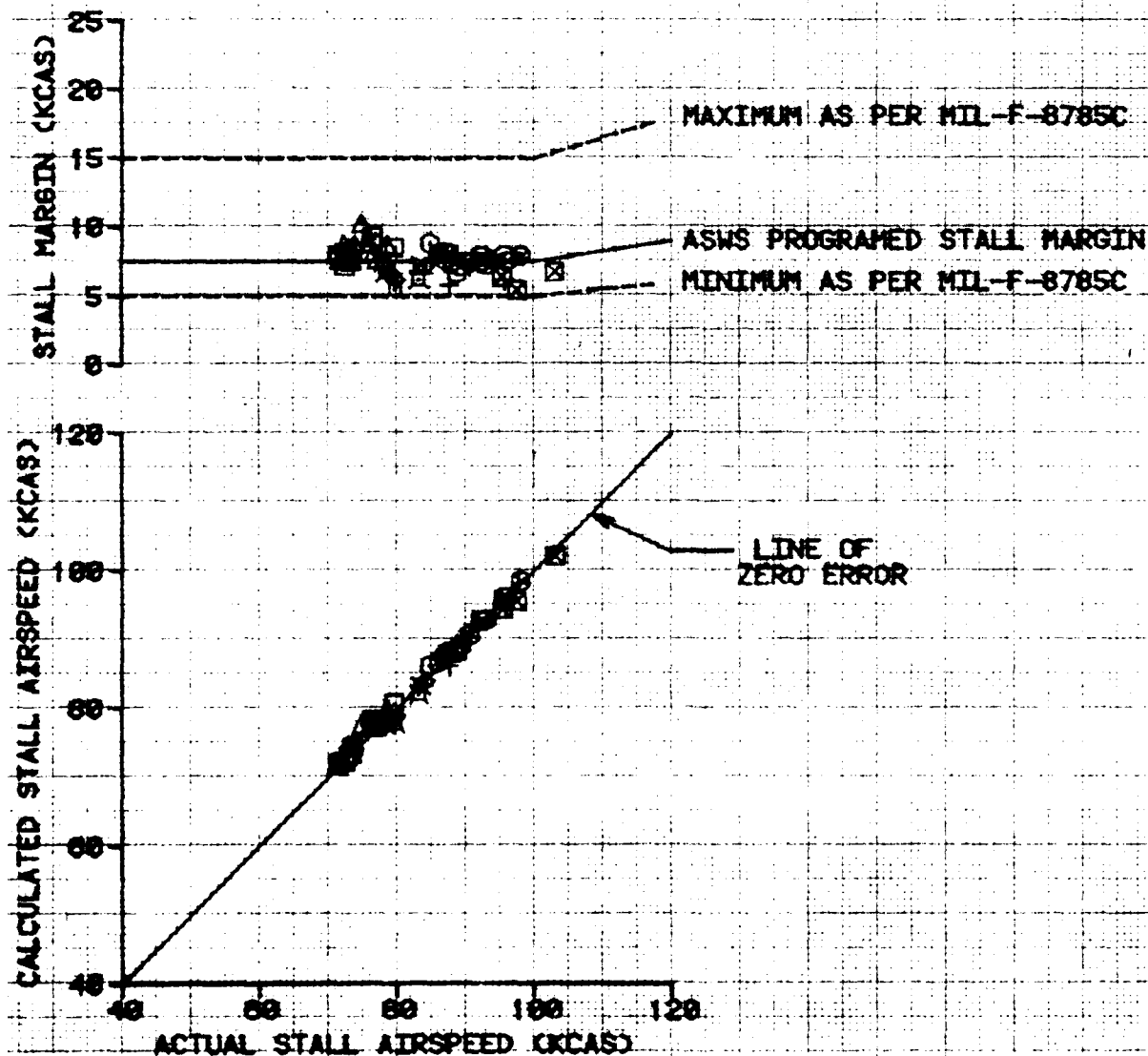BETWEEN THE STALL WARNING AIRSPEED AND THE
ACTUAL STALL AIRSPEED

MAXIMUM AS PER MIL-F-8785C

ASWS PROGRAMED STALL MARGIN

MINIMUM AS PER MIL-F-8785C

STALL MARGIN (KCAS)

LINE OF ZERO ERROR

CALCULATED STALL AIRSPEED (KCAS)

ACTUAL STALL AIRSPEED (KCAS)

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FIGURE 5
AUTOMATED STALL WARNING SYSTEM
STALL AIRSPEED SUMMARY
15 DEGREES FLAPS
JOV-1C USA S/N 68-03748

AVERAGE GROSS WEIGHT (LB)	AVERAGE LONGITUDINAL CG LOCATION (FS)	AVERAGE DENSITY ALTITUDE (FEET)	AVERAGE VERTICAL ACCELERATION (G)	PROPELLER SPEED (RPM)	LANDING GEAR POSITION
13540	160.1	11470	1.0	1678	EXTENDED
13060	159.9	11140	1.5	1678	EXTENDED
13570	160.1	11430	1.0	1600	RETRACTED

NOTE: STALL MARGIN IS DEFINED AS THE DIFFERENCE BETWEEN THE STALL WARNING AIRSPEED AND THE ACTUAL STALL AIRSPEED
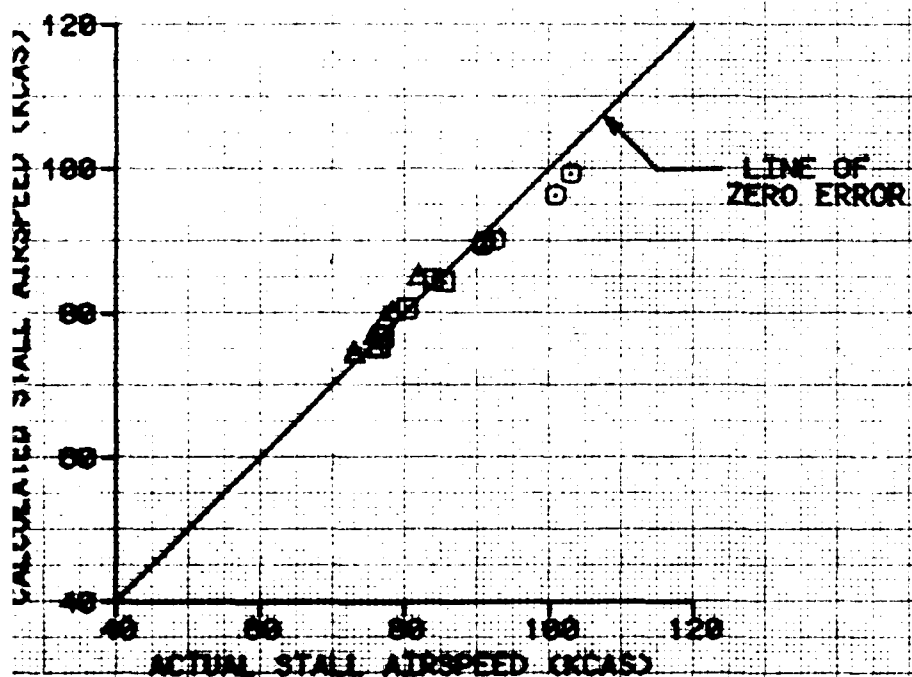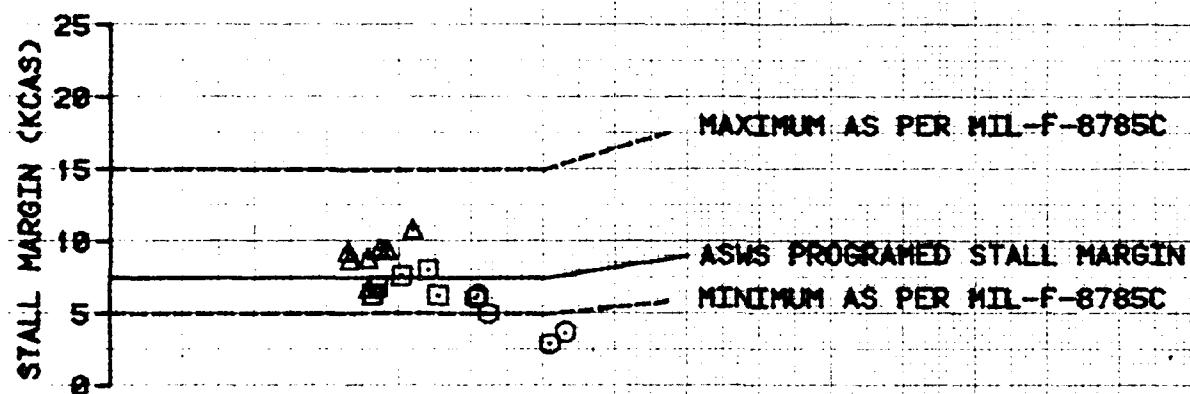

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FIGURE 4
AUTOMATED STALL WARNING SYSTEM
STALL AIRSPEED SUMMARY
ZERO DEGREES FLAPS
JOV-1C USA S/N 60-03748

SYM	AVERAGE GROSS WEIGHT (LB)	AVERAGE LONGITUDINAL CG LOCATION (FS)	AVERAGE DENSITY ALTITUDE (FEET)	AVERAGE VERTICAL ACCELERATION (G)	PROPELLER SPEED (RPM)	LANDING GEAR POSITION
▪	12000	157.0	11500	1.0	1400	RETRACTED
⊟	12480	158.9	10000	1.9	1400	RETRACTED
▲	13150	160.0	11620	1.0	1600	EXTENDED
+	13040	160.1	11570	1.0	1400	RETRACTED
◆	13430	160.0	11400	1.9	1400	RETRACTED
★	13670	160.1	14120	1.0	1670	RETRACTED
⊠	15780	160.1	11470	1.0	1400	RETRACTED
⊞	15420	160.0	11920	1.8	1400	RETRACTED
■	16610	166.0	11870	1.0	1400	RETRACTED
▼	15740	166.2	11500	1.0	1400	RETRACTED
■	12550	159.0	10550	2.7	1400	RETRACTED

NOTES: 1. DATA REPRESENTED BY ★ DENOTES SIMULATED SINGLE ENGINE CONDITION (LEFT ENGINE AT FLIGHT IDLE, MINIMUM RPM)
2. STALL MARGIN IS DEFINED AS THE DIFFERENCE BETWEEN THE STALL WARNING AIRSPEED AND THE ACTUAL STALL AIRSPEED
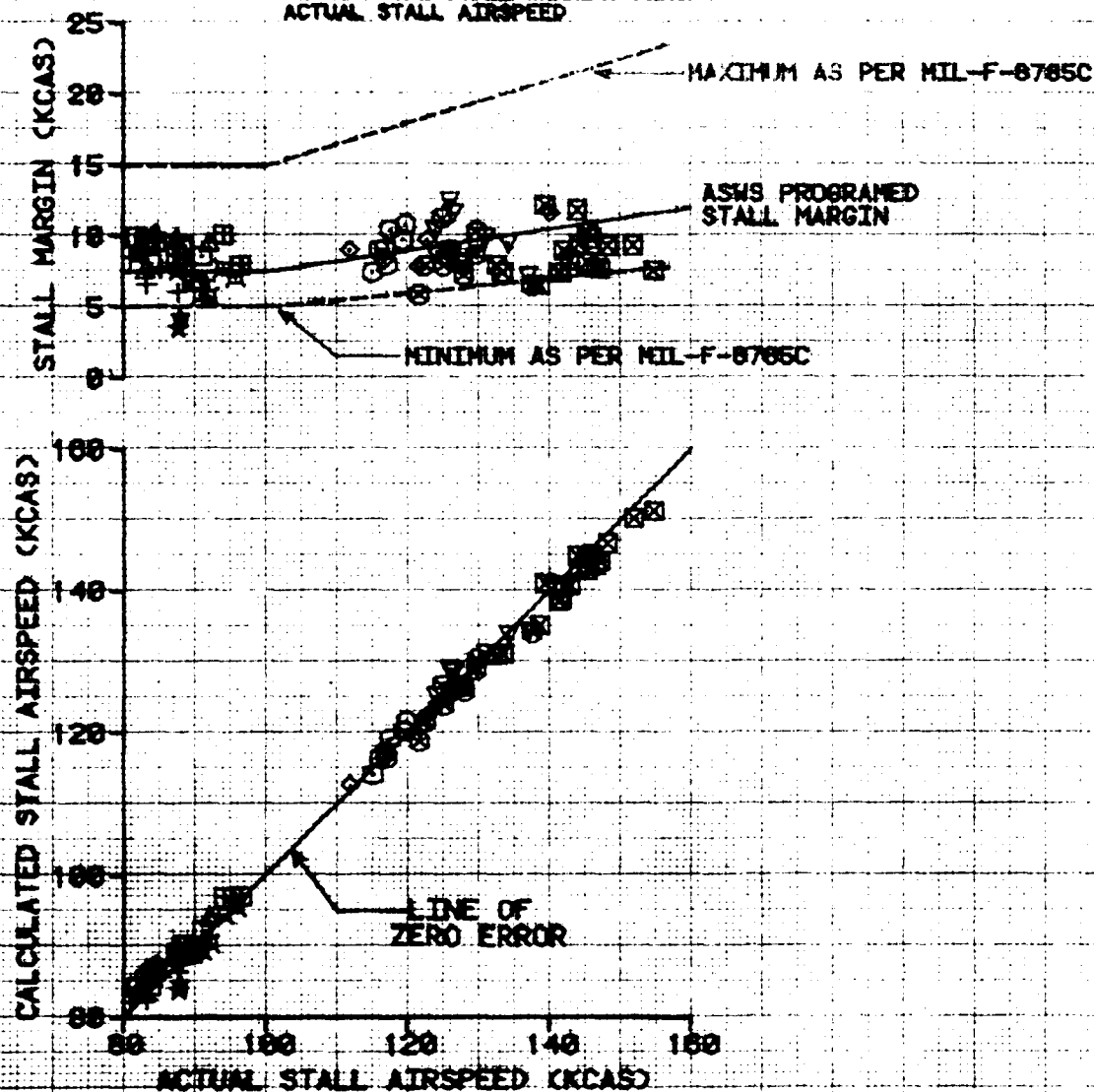
FIGURE 3
AUTOMATED STALL WARNING SYSTEM
SOFTWARE COEFFICIENT DETERMINATION
45 DEGREES FLAPS
JOV-1C USA S/N 60-03748

SYM	AVERAGE GROSS WEIGHT (LB)	AVERAGE LONGITUDINAL CG LOCATION (FS)	AVERAGE DENSITY ALTITUDE (FEET)	PROPELLER SPEED (RPM)	LANDING GEAR POSITION
⊙	12750	157.0	10810	1600	EXTENDED
⊡	13460	160.0	13910	1600	EXTENDED
△	15710	160.1	11590	1600	EXTENDED
+	16010	166.1	11620	1600	EXTENDED

SECOND-ORDER POLYNOMIAL COEFFICIENTS

A = 1.894037

B = 0.1327434E-01

C = -0.1155807E-03

$$Y = A + BX + CX^2$$

AIRCRAFT COEFFICIENT OF LIFT

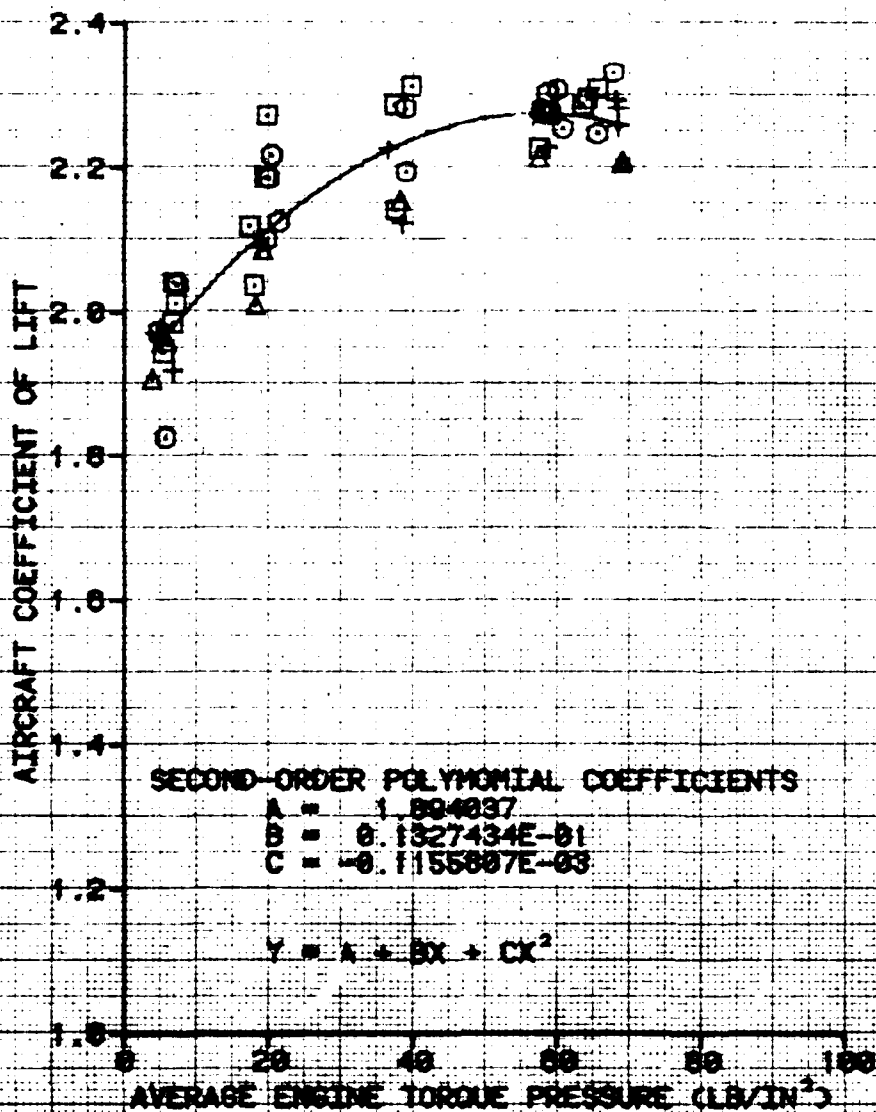AVERAGE ENGINE TORQUE PRESSURE (LB/IN^2)

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FIGURE 2
AUTOMATED STALL WARNING SYSTEM
SOFTWARE COEFFICIENT DETERMINATION
15 DEGREES FLAPS
JOV-1C USA S/N 60-03748

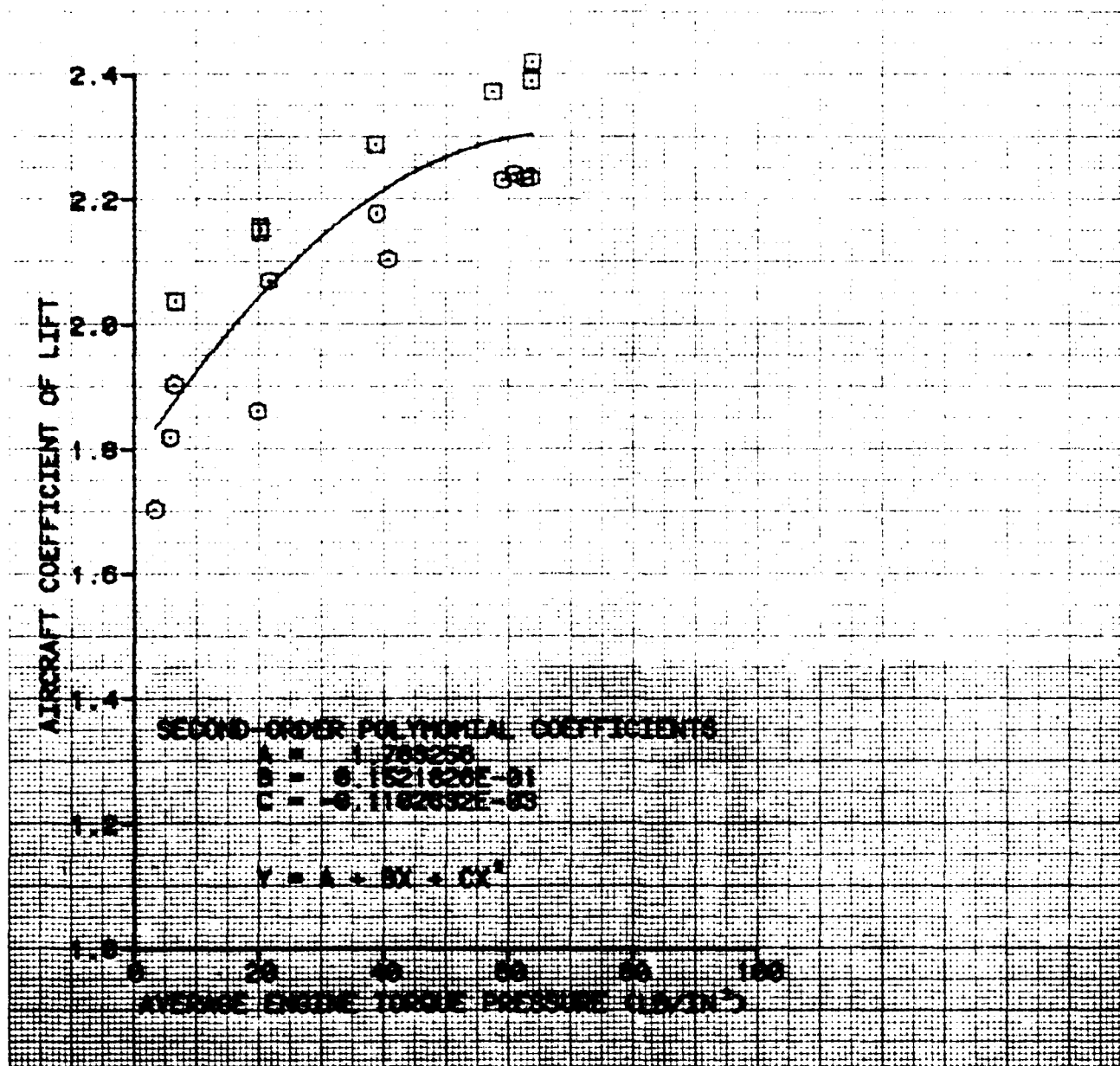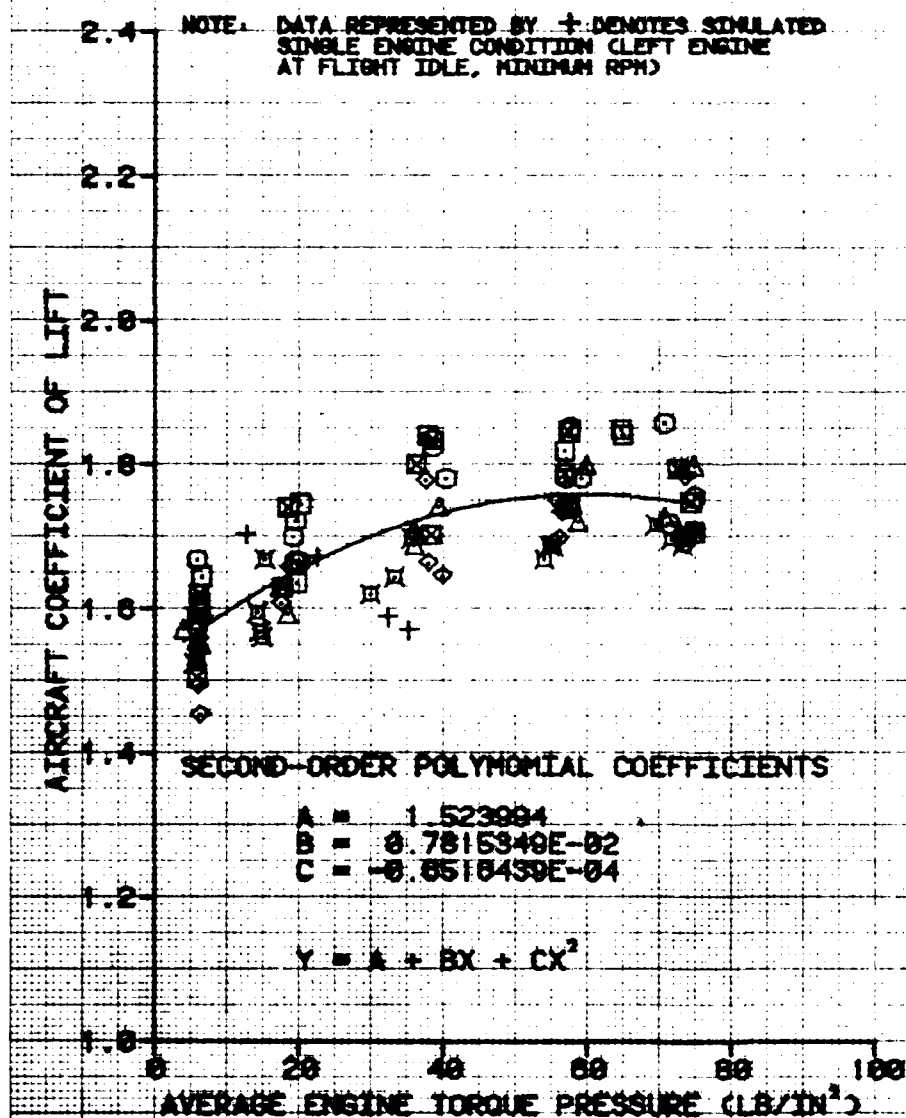| SYM	AVERAGE GROSS WEIGHT (LB)	AVERAGE LONGITUDINAL CG LOCATION (FS)	AVERAGE DENSITY ALTITUDE (FEET)	PROPELLER SPEED (RPM)	LANDING GEAR POSITION
⊙	13300	160.0	11300	1678	EXTENDED
⊡	13570	160.1	11430	1600	RETRACTED

FIGURE 1
AUTOMATED STALL WARNING SYSTEM
SOFTWARE COEFFICIENT DETERMINATION
ZERO DEGREES FLAPS
JOV-1C USA S/N 60-03748

SYM	AVERAGE GROSS WEIGHT (LB)	AVERAGE LONGITUDINAL CG LOCATION (FS)	AVERAGE DENSITY ALTITUDE (FEET)	PROPELLER SPEED (RPM)	LANDING GEAR POSITION
⊙	12720	157.0	10800	1400	RETRACTED
⊡	13130	160.0	11620	1600	EXTENDED
△	13540	160.1	11400	1400	RETRACTED
+	13670	160.1	14120	1678	RETRACTED
◇	15800	160.1	11700	1400	RETRACTED
▧	15930	166.1	11730	1400	RETRACTED
✕	12550	159.0	10550	1400	RETRACTED

NOTE: DATA REPRESENTED BY + DENOTES SIMULATED
SINGLE ENGINE CONDITION (LEFT ENGINE
AT FLIGHT IDLE, MINIMUM RPM)

SECOND-ORDER POLYNOMIAL COEFFICIENTS

A = 1.523994
B = 0.7815349E-02
C = -0.6516439E-04

$Y = A + BX + CX^2$

AIRCRAFT COEFFICIENT OF LIFT

AVERAGE ENGINE TORQUE PRESSURE (LB/IN^2)

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APPENDIX E.

Index

airspeed (V_{cal}) was obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic} and position error (ΔV_{pc}).

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc} \qquad (12)$$

Rigging Check

5. Mechanical rigging of engine and flight controls was checked for compliance with applicable technical manuals and Lycoming documents.

Lift from the tail (L_T), lb

$$L_T = Q \cdot S_T [0.78 - 0.91 C_T + 0.78 (C_T)^2 \\ - 0.0147 \, \delta_f + 1.06 \times 10^{-4} (\delta_f)^2 + 0.248 \, \delta_e] \qquad (9)$$

where:

S_T = Tail area = 85 ft^2
δ_f = Flap position (degrees)
δ_e = Elevator position (degrees)

The coefficients of equation 9 are based on tail coefficient of lift corrected for changes in effective angle-of-attack of the tail due to power and flap position influences. Corrections derived from the data of Grumman report number GWTT 249, November 1967.

Coefficient of Lift at Stall C_{Lmax}

The value of CL_{max} is a function of flap angle and engine thrust.

$$CL_{max} = C_L A \, (Q_{avg})^2 + C_L B \, (Q_{avg}) + C_L C \qquad (10)$$

The values of the coefficients ($C_L A$, $C_L B$, $C_L C$) are dependent on the flap position selected. Values of C_{Lmax} are determined experimentally through equation 1, where all other values can be measured or calculated. Rearrangement of equation 1 results in the following equation.

$$C_{Lmax} = \frac{.702113 \, [GW(n_x \sin(\alpha) + n_z \cos(\alpha)) + L_T - T_z]}{\rho_o \cdot S_W \cdot V_{stall}^2} \qquad (11)$$

Using equation 11 and measured engine torque values, the coefficients of equation 10 can be determined using a 2nd-order polynomial regression. Three sets of these coefficients are required for the three flap settings used, with each set optimized for various gross weight and center of gravity conditions. Data sets utilizing the above technique and their associated coefficients are shown in figures 1 through 3, appendix E.

Airspeed Calibration

4. The test boom, ASWS and the ship's standard pitot-static system were calibrated using the pace aircraft method to determine the airspeed position error (figs. 7 through 12). Calibrated

where.

$$P_a = \text{Ambient Pressure (in.-Hg)}$$

True airspeed (V_T), ft/sec

$$V_T = V_E \ (\rho_o/\rho)^{1/2} \qquad\qquad (4)$$

where:

$$V_E = \text{Equivalent Airspeed} = V_{cal} \ (\text{i.e., assume} \qquad (5)$$
$$\text{incompressible flow})$$

$$V_{CAL} = V_{IND} + \Delta V_{pc} \ (\text{position error correction})$$

$$\rho = 0.0023769 - 6.84 \times 10^{-5} \ H_P \ (\text{assume standard day}$$
$$+ 6.4 \times 10^{-7} \ H_P^2 \ \text{slugs/ft}^3 \qquad \text{temperature})$$

Total Thrust (T), lb, (Velocity >60 KTAS)

$$T = 744.0 + V_T \ (0.00642 \ V_T$$
$$- 0.0452 \ Q_{avg} - 4.72) + Hp \ (1.219 \qquad (6)$$
$$+ 0.004 \ V_T - 0.7969 \ Q_{avg}$$
$$+ 0.00122 \ V_T \cdot Q_{avg}) + 32.666 \ Q_{avg}$$

where:

$$Q_{avg} = \text{Average engine torque}$$
$$= 0.5 \ (Q_L + Q_R)$$

This equation was derived from the thrust/velocity plots of Grumman report number XA 134-105-13, figures III-12 through III-17.

Vertical component of thrust (T_z), lb

$$T_z = (T) \sin (\alpha + C\phi) \qquad\qquad (7)$$

where:

$$C_\phi = \text{Engine thrust angle} = 1.92 \text{ degrees}$$

Thrust coefficient (C_T)

$$C_T = T_z / ((SIN(\alpha + C_\phi) \cdot Q \cdot S_W)) \qquad\qquad (8)$$

S_W = Wing area = 330 ft^2
C_{Lmax} = Aircraft coefficient of lift at stall
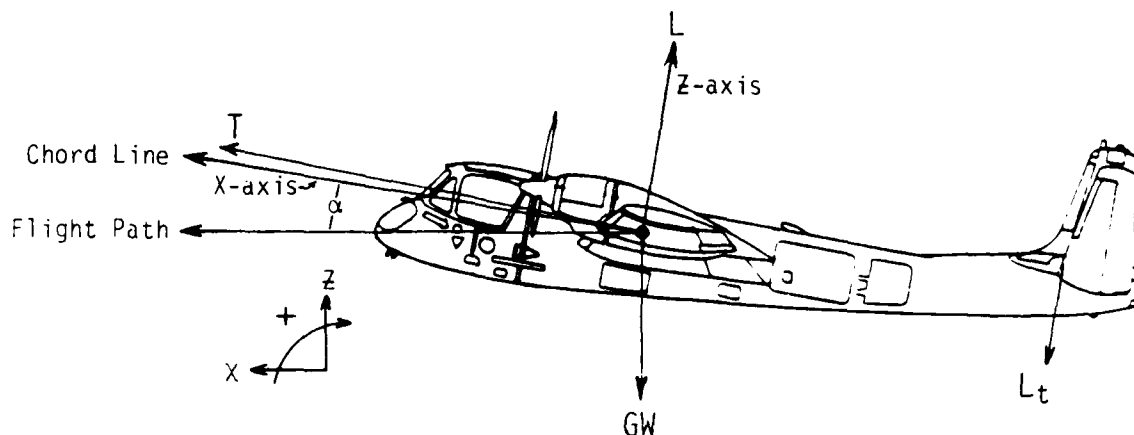.5925 = Conversion factor knots/ft/sec

Figure 1. OV-1 Free-Body Diagram

Input values for this equation are determined from aircraft sensors and the following NASA/DFRF derived equations.

<u>Indicated airspeed (V_{IND}), ft/sec</u>

$$V_{IND} = \left[\frac{2\ (70.7262)\ \text{lb/ft}^2/\text{in.-Hg} \cdot Q}{\rho_0} \right]^{1/2} \qquad (2)$$

where.

    Q = Dynamic pressure (in.-Hg)
    70.7262 = Conversion factor

<u>Pressure Altitude H_p, 1000 ft</u>

$$H_p = 145.442 \left[1 - \left| \frac{P_a}{29.9213} \right|^{0.190262} \right] \qquad (3)$$

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APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. Stall data were obtained in ball-centered flight at the conditions shown in table 2. Unaccelerated stalls were conducted wings level at 1 kt/sec or less deceleration; accelerated stalls were conducted using windup turns at constant load factor with a deceleration of 2 kt/sec or less. Data records were taken just prior to stall, with stall being defined as the point at which uncommanded pitching, rolling or yawing occurs, or intolerable buffet or structural vibration is encountered (MIL-F-8785C, para 6.2.2 B&C). Test flights were flown at a pressure altitude of 10,000 feet with a longitudinal center of gravity (cg) location maintained near the proposed mission value. In addition, cg shift due to fuel burn varied by less than 0.20 inches. Data obtained were analyzed to determine the coefficients required to drive the stall warning system within the limits of MIL-F-8785C (ref 6, app A).

Aircraft Weight and Balance

2. The test aircraft was weighed at the start of the test program with all instrumentation installed, full oil and full internal fuel. Drop tanks were installed at wing stations 3 and 4 and were empty. The basic aircraft weight minus fuel (1900.8 lb) and oil (38 lb) was determined to be 11,088 pounds with a longitudinal cg location at fuselage station 163.40 and a lateral cg location at buttline 0.0.

Stall Equation Coefficients

3. The automated stall warning system (ASWS) cockpit indicator stall airspeed is determined using the following equation, based on the conventions of figure 1:

$$V_{stall} = 0.5925 \left[\frac{2 \; GW \, [\sin(\alpha)\,(n_x) + \cos(\alpha)\,(n_z)] + L_T - T_z}{\rho_0 \cdot S_W \cdot C_{L_{max}}} \right]^{1/2} \quad \text{Knots Calibrated Airspeed} \quad (1)$$

Where:

GW = Aircraft weight (pounds)
α = Stall angle-of-attack = 21.7 degrees*
n_x = Horizontal load factor (g)
n_z = Vertical load factor (g)
L_T = Lift from the tail (pounds)
T_z = Vertical component of thrust (pounds)
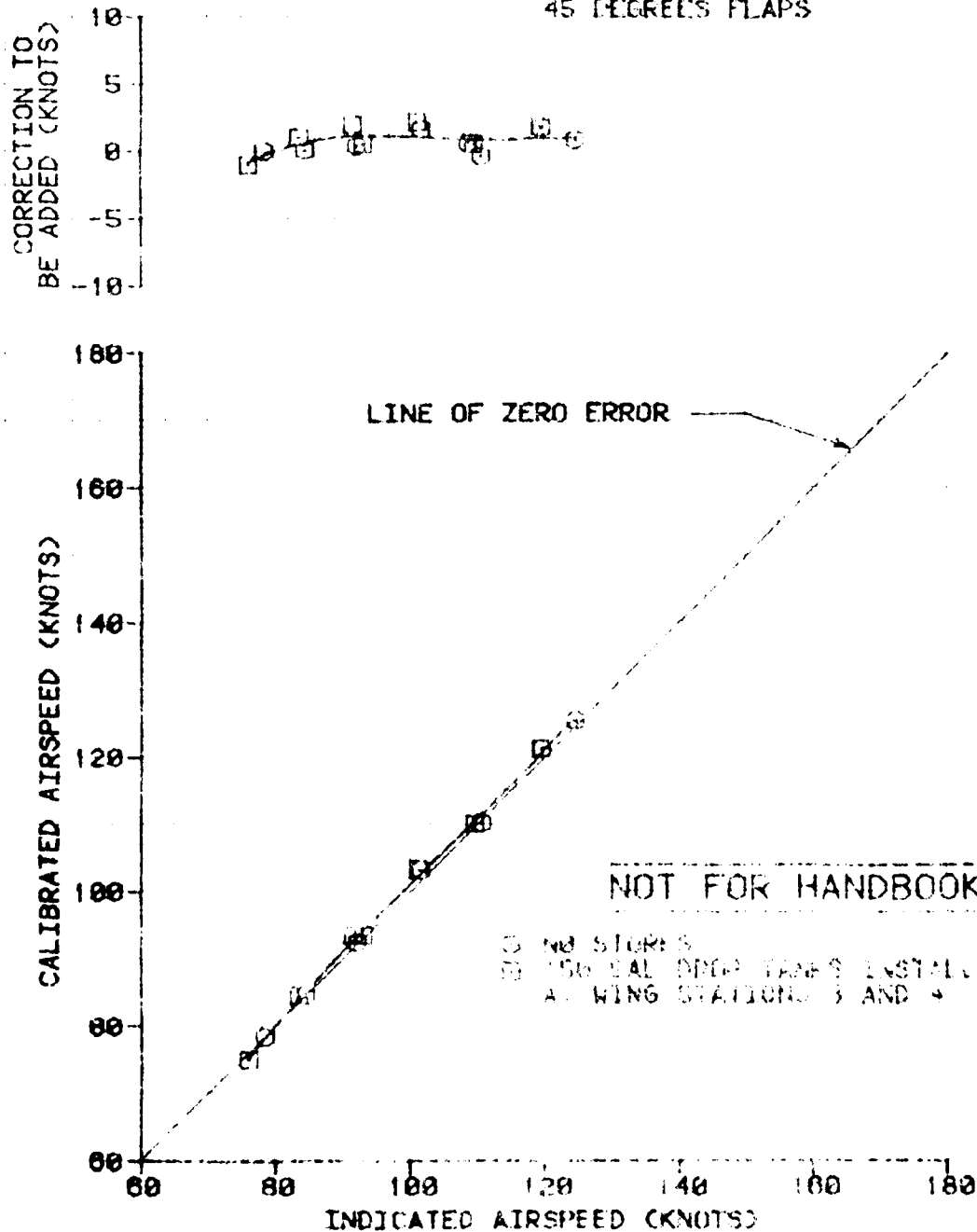ρ_0 = 0.0023769 slugs/ft^3

*As determined by Grumman Aircraft from wind tunnel data

Figure

BOOM AIRSPEED CALIBRATION
JOV-1C USA S/N 60-3748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION		AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AIRCRAFT CONFIGURATON
		LONG (FS)	LAT (BL)			
☐	12080	159.5	0.0	9870	18 5	LANDING
☐	12400	159.8	0 0	9100	7 9	LANDING

NOTE: 1. T-28 PACE METHOD

2. LANDING GEAR EXTENDED,
45 DEGREES FLAPS

LINE OF ZERO ERROR

NOT FOR HANDBOOK USE

☐ NO STORES
☐ 150 GAL DROP TANKS INSTALLED
AT WING STATIONS 3 AND 4

FIGURE 2
BOOM AIRSPEED CALIBRATION
JOV-1C USA S/N 60-3748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AIRCRAFT CONFIGURATON
⊙	12400	159.6	0.0	9940	11.0	SLOW FLIGHT
⊡	12800	159.9	0.0	9200	7.5	SLOW FLIGHT

NOTE: 1. T-28 PACE METHOD

2. LANDING GEAR EXTENDED, 15 DEGREES FLAPS

LINE OF ZERO ERROR

NOT FOR HANDBOOK USE

⊙ NO STORES
⊡ 150 GAL DROP TANKS INSTALLED AT WING STATIONS 3 AND 4

CORRECTION TO BE ADDED (KNOTS)

CALIBRATED AIRSPEED (KNOTS)

INDICATED AIRSPEED (KNOTS)

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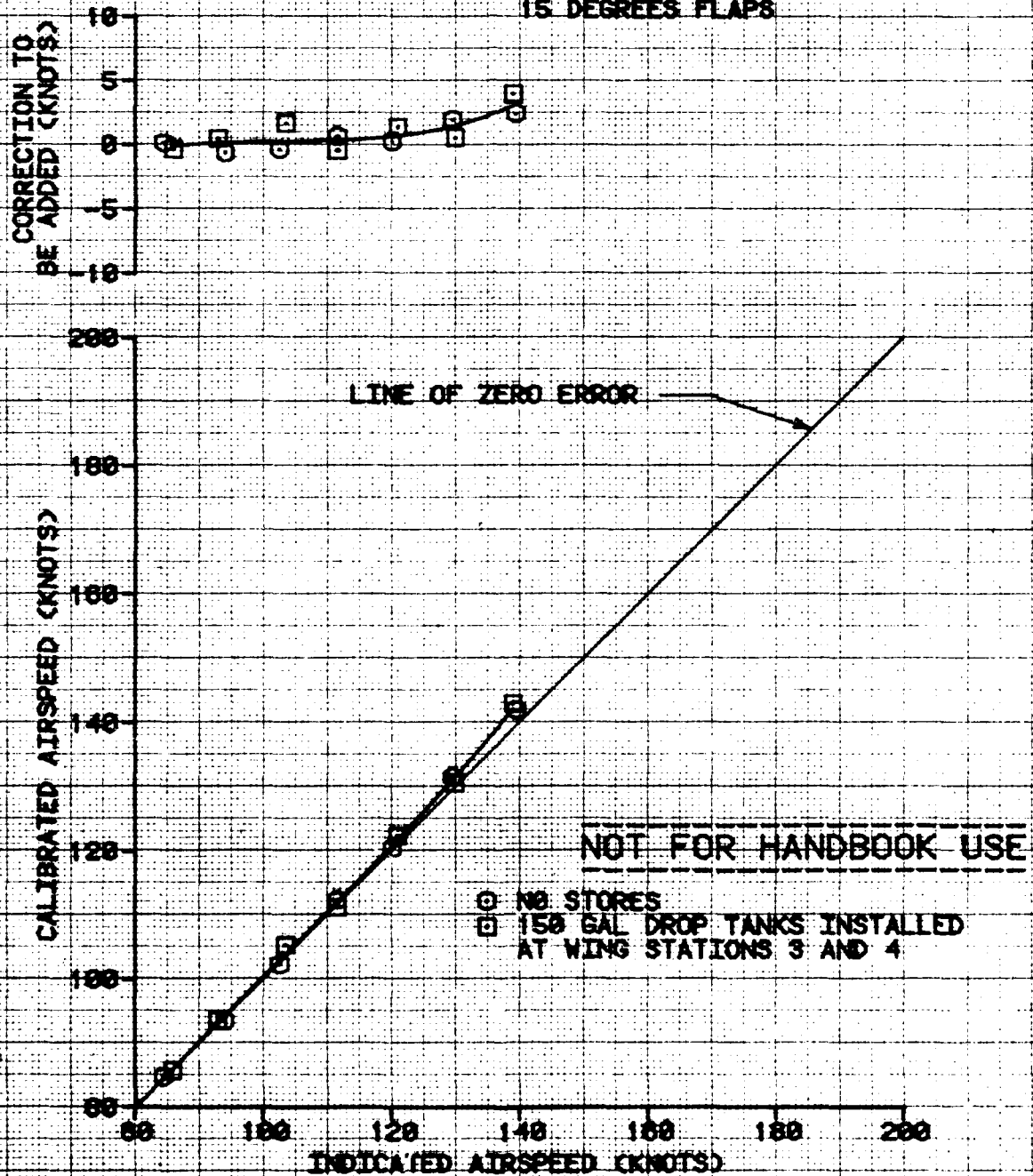
FIGURE 7
SHIP SYSTEM AIRSPEED CALIBRATION
JOV-1C USA S/N 60-3748

| SYM | AVG GROSS WEIGHT (LB) | AVG CG LOCATION | | AVG DENSITY ALTITUDE (FT) | AVG OAT (DEG C) | AIRCRAFT CONFIGURATION |
		LONG (FS)	LAT (BL)			
◎	12910	159.7	0.0	10040	12.0	CRUISE
⊡	13200	159.9	0.0	9420	9.5	CRUISE

NOTE: 1. T-28 PACE METHOD

2. LANDING GEAR RETRACTED, ZERO DEGREES FLAPS

LINE OF ZERO ERROR

◎ NO STORES
⊡ 150 GAL DROP TANKS INSTALLED AT WING STATIONS 3 AND 4

CORRECTION TO BE ADDED (KNOTS)

CALIBRATED AIRSPEED (KNOTS)
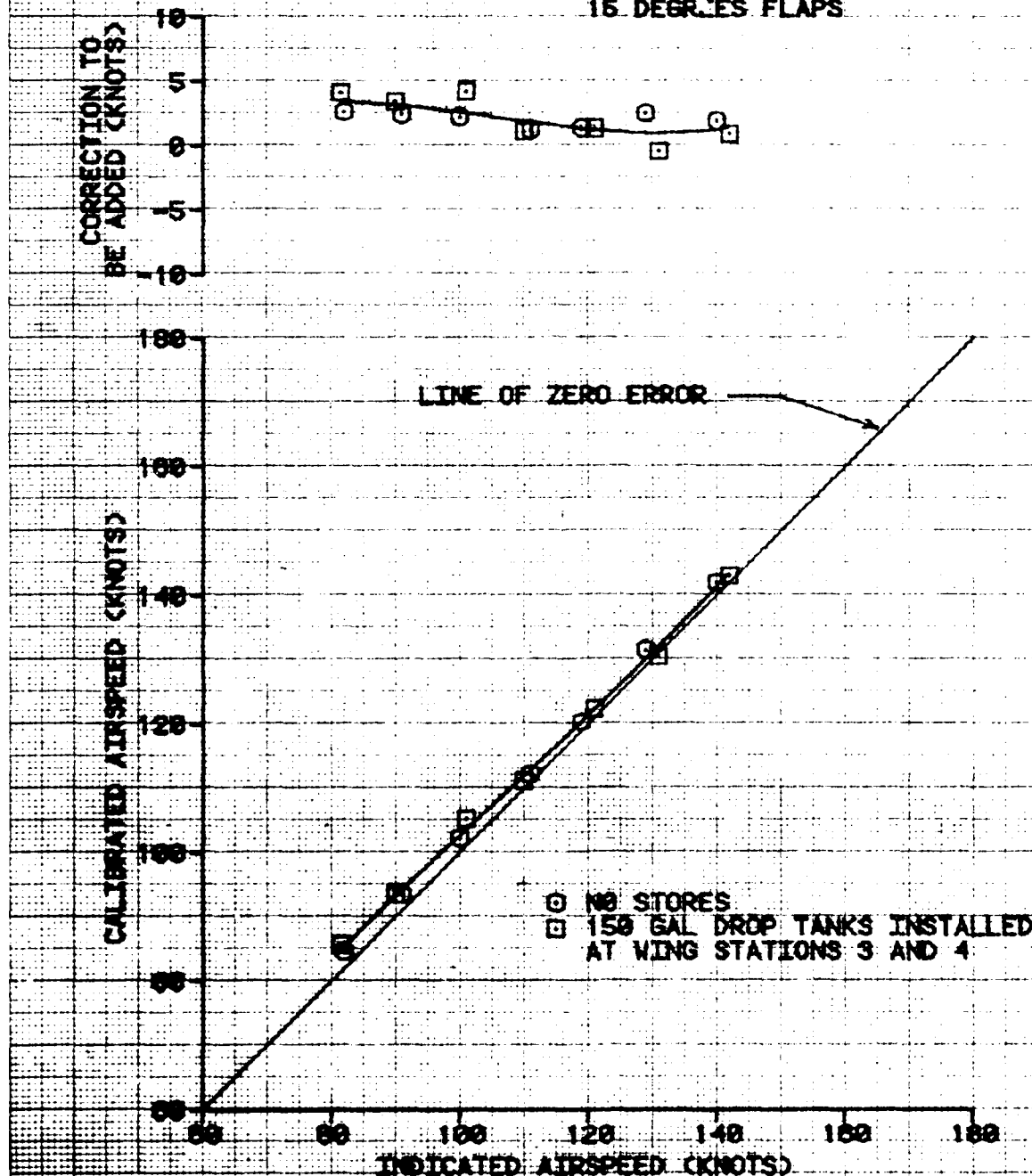
INDICATED AIRSPEED (KNOTS)

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FIGURE 8
SHIP SYSTEM AIRSPEED CALIBRATION
JOV-1C USA S/N 69-03748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION		AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AIRCRAFT CONFIGURATON
		LONG (FS)	LAT (BL)			
⊙	12400	159.6	0.0	9940	11.0	SLOW FLIGHT
⊡	12800	159.9	0.0	9200	7.5	SLOW FLIGHT

NOTE: 1. T-28 PACE METHOD.

2. LANDING GEAR EXTENDED, 15 DEGREES FLAPS

LINE OF ZERO ERROR

⊙ NO STORES
⊡ 150 GAL DROP TANKS INSTALLED AT WING STATIONS 3 AND 4

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FIGURE 9
SHIP SYSTEM AIRSPEED CALIBRATION
JOV-1C USA S/N 60-03748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION		AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AIRCRAFT CONFIGURATON
		LONG (FS)	LAT (BL)			
⊙	12080	159.5	0.0	9870	10.5	LANDING
⊡	12400	159.8	0.0	9180	7.0	LANDING

NOTE: 1. T-28 PACE METHOD

2. LANDING GEAR EXTENDED, 45 DEGREES FLAPS

LINE OF ZERO ERROR

⊙ NO STORES
⊡ 150 GAL DROP TANKS INSTALLED AT WING STATIONS 3 AND 4

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FIGURE 10
AUTOMATED STALL WARNING SYSTEM
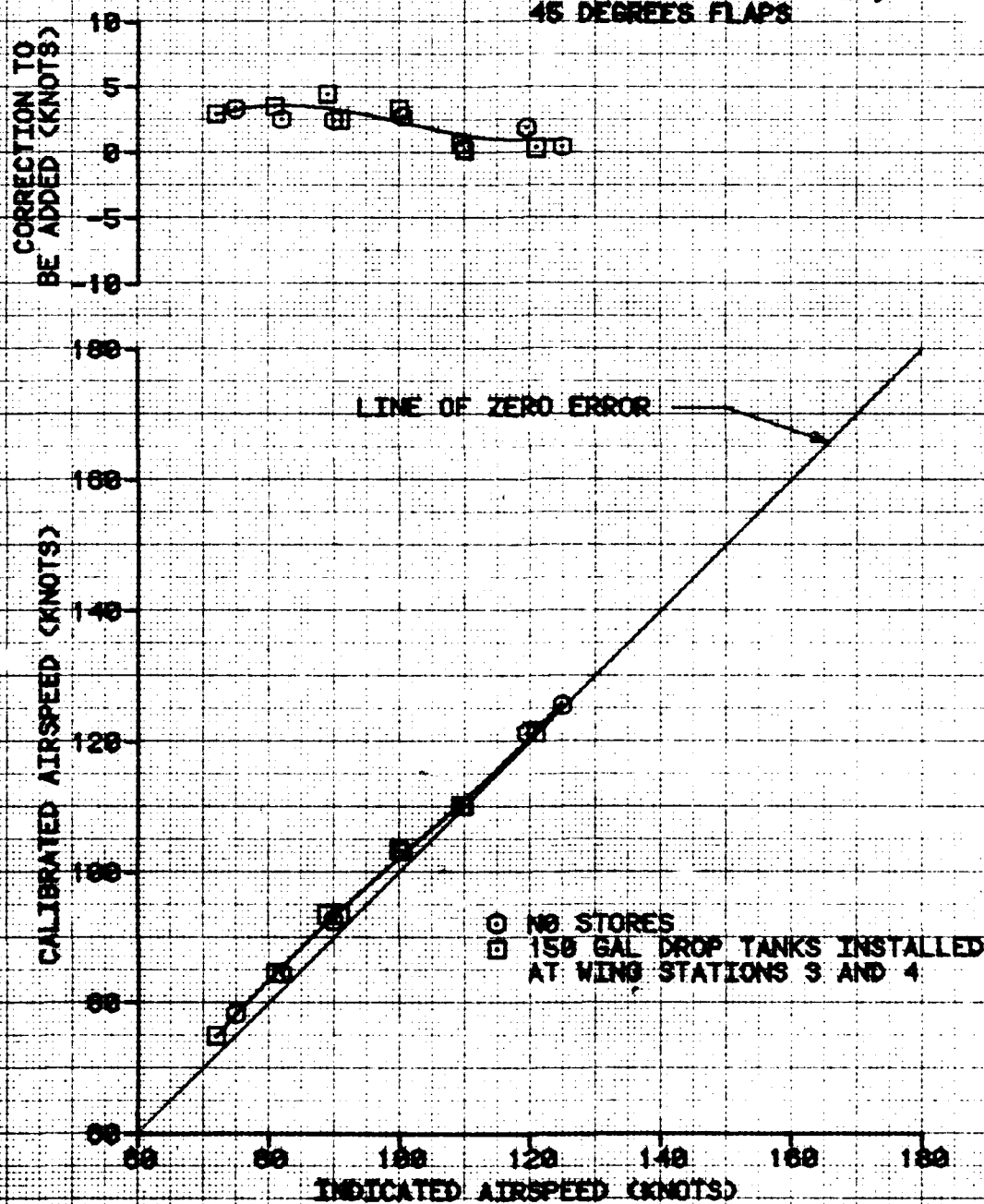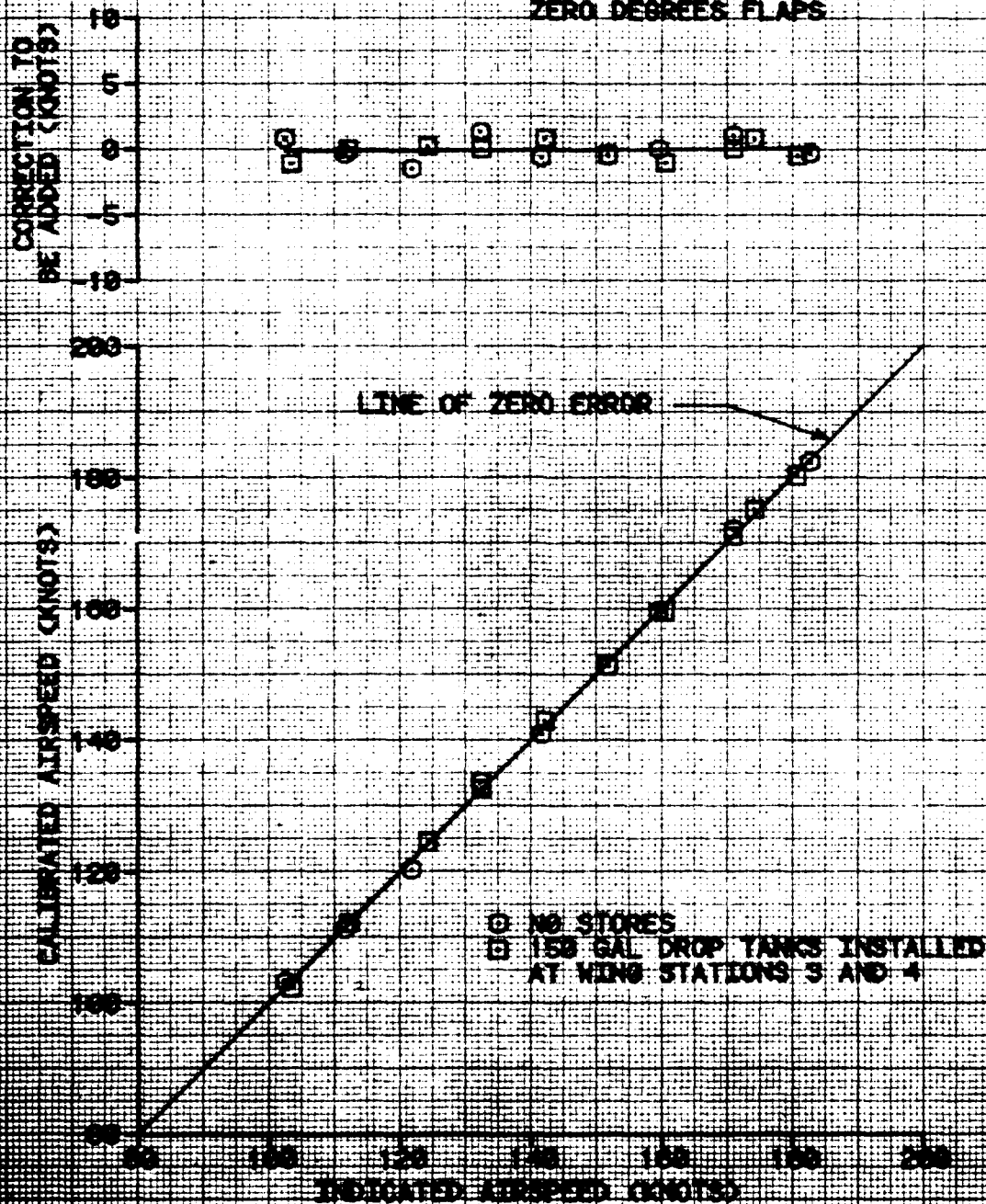AIRSPEED CALIBRATION
JOV-1C USA S/N 60-3748

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FIGURE 11
AUTOMATED STALL WARNING SYSTEM
AIRSPEED CALIBRATION
JOV-1C USA S/N 80-3748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AIRCRAFT CONFIGURATON
⊙	12400	159.8	0.0	9940	11.0	SLOW FLIGHT
⊡	12800	160.9	0.0	9200	7.5	SLOW FLIGHT

NOTE: 1. T-28 PACE METHOD

2. LANDING GEAR EXTENDED, 15 DEGREES FLAPS

LINE OF ZERO ERROR

⊙ NO STORES
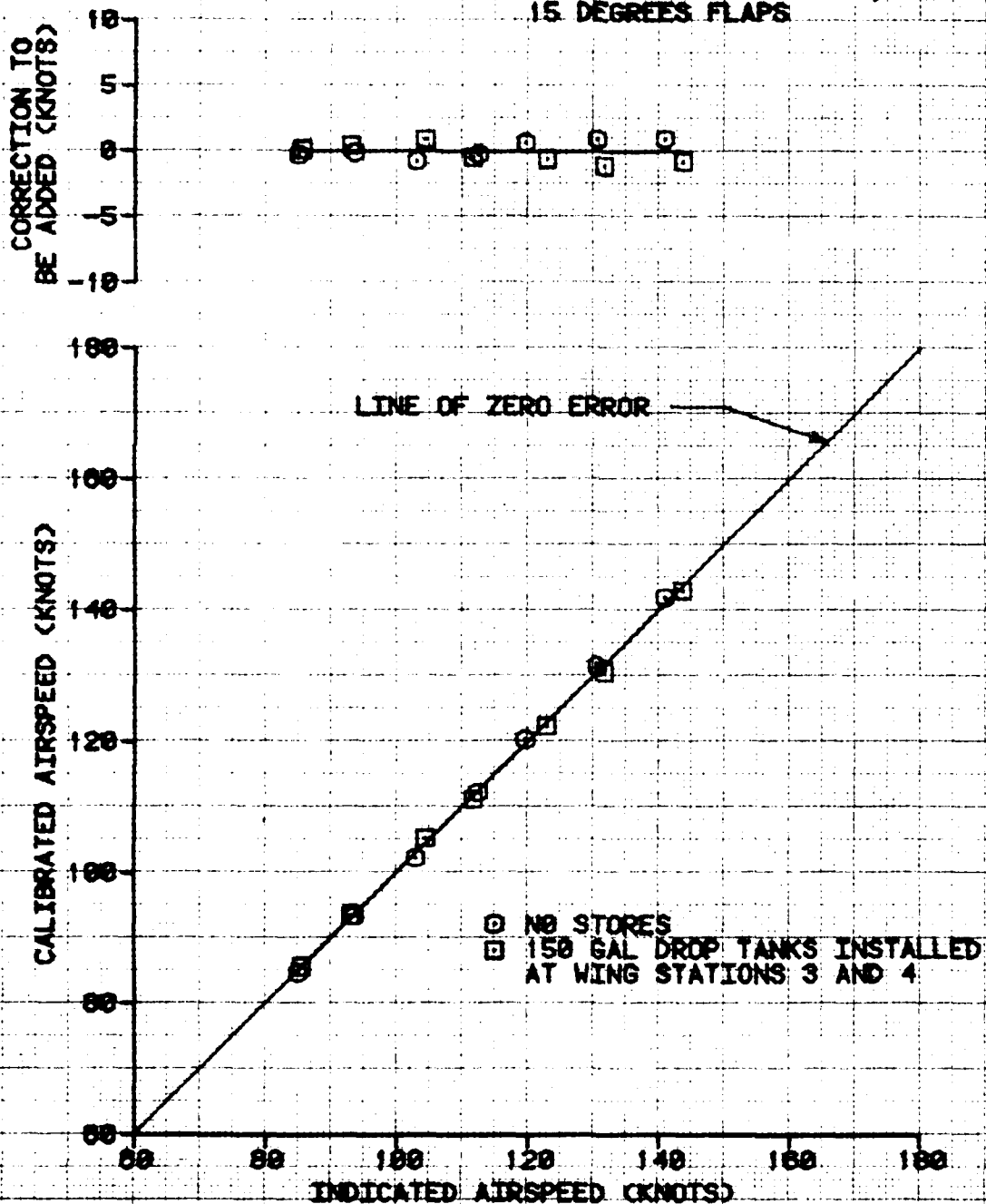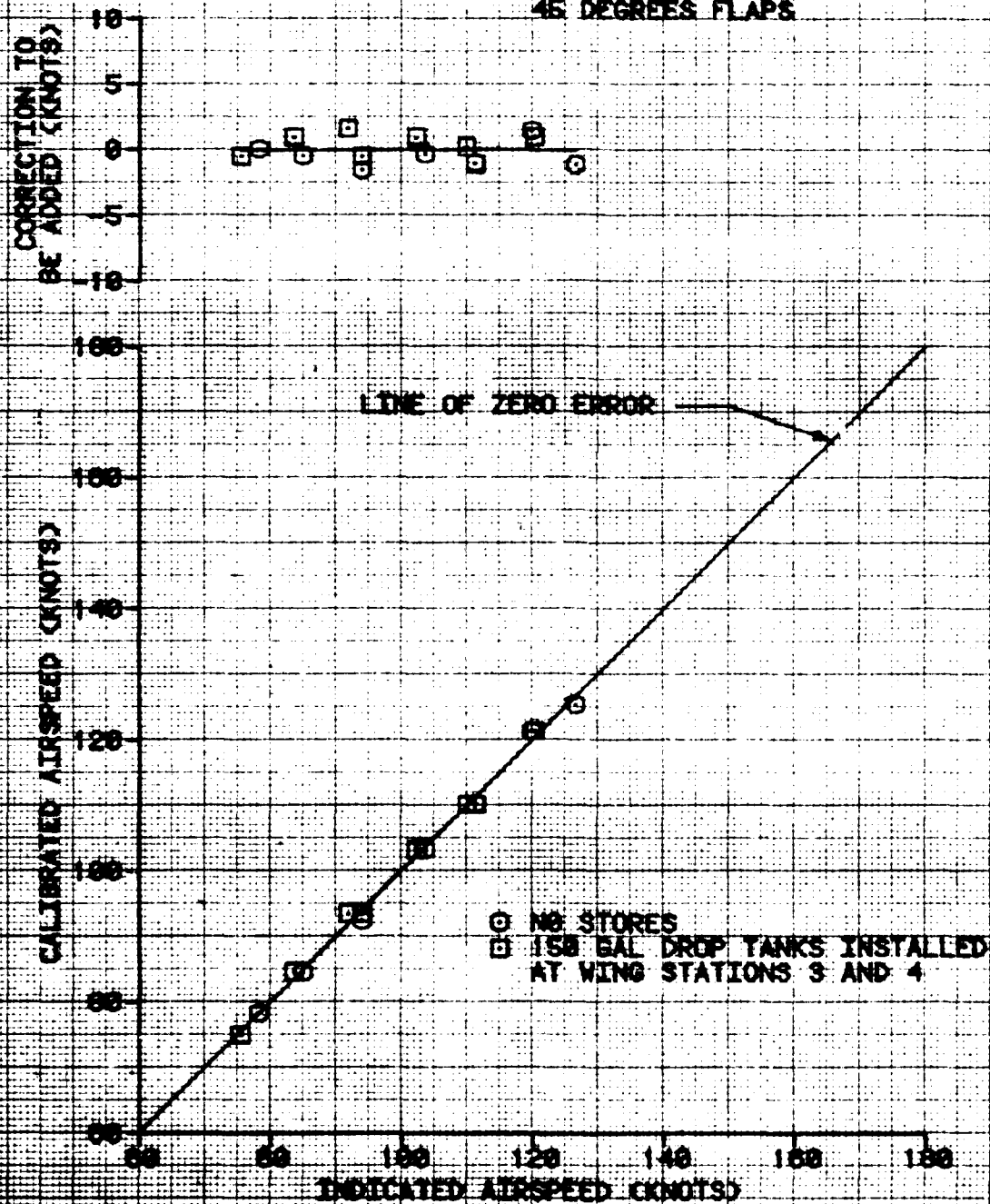⊡ 150 GAL DROP TANKS INSTALLED AT WING STATIONS 3 AND 4

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FIGURE 12
AUTOMATED STALL WARNING SYSTEM
AIRSPEED CALIBRATION
JOV-1C USA S/N 68-3748

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION		AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG. C)	AIRCRAFT CONFIGURATON
		LONG (FS)	LAT (BL)			
⊙	12000	159.5	0.0	9870	10.5	LANDING
⊡	12400	160.0	0.0	9180	7.0	LANDING

NOTE: 1. T-28 PACE METHOD
2. LANDING GEAR EXTENDED, 45 DEGREES FLAPS

LINE OF ZERO ERROR

⊙ NO STORES
⊡ 150 GAL DROP TANKS INSTALLED AT WING STATIONS 3 AND 4

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